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LUNAR LANDER PROJECT:

A STUDY ON FUTURE MANNED MISSIONS TO THE MOON

(NASA-CR-197177) LUNAR LANDER PROJECT: A STUDY ON FUTURE MANNED MISSIONS TO THE MOON (USRA) 77 p

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ABSTRACT

This project is based on designing a small lunar probe which will conduct research relating to future manned missions to the moon. The basic design calls for two experiments to be run. The first of these experiments is an enclosed environment section which will be exposed to solar radiation while on the moon. The purpose of this experiment is to determine the effect of radiation on an enclosed environment and how different shielding materials can be used to moderate this effect. The eight compartments will have the following covering materials: glass, polarized glass, plexiglass, polyurethane, and boron impregnated versions of the polyutethane and plexiglass. The enclosed atmosphere will be sampled by a mass spectrometer to determine elemental breakdown of its primary constituents. This is needed so that an accurate atmospheric processing system can be designed for a manned mission. The second experiment is a seismic study of the moon. A small penetrating probe will be shot into the lunar surface and data will be collected onboard the lander by an electronic seismograph which will store the data in the data storage unit for retrieval and transmission once every twenty-three hours.

The project is designed to last ten years with possible extended life for an additional nine years at which point power requirements prevent proper functioning of the various systems.

LUNAR LANDER INITIAL DESIGN REQUIREMENTS

GROUP MEMBERS

1/C FRANK GIANOCARO	- TEAM LEADER - THERMAL CONTROLS - EXPERIMENTAL PAYLOAD
1/C MICHAEL CARTER	- ATTITUDE DYNAMICS / CONTROLS
1/C NOEL FAGAN	- STRUCTURES - MECHANISMS
1/C BRIAN HAWKINS	- COMMUNICATIONS
1/C TODD HUBER	- PROPULSION
1/C RICHARD RIVERA	- POWER
1/C JAY WOODRUFF	- ORBITAL-MECHANICS

- Plan for launch sometime in 1998.

- Launch vehicle to be used will be a Delta II 7920/25.

- Provide consistent power over a ten year period (mission life).

- Must make a soft landing on the lunar surface to protect sensitive instruments on board spacecraft from decalibration due to excessive impact forces.

- Primary lander mission will be to study the lunar environment for follow-on manned missions. Specifically, this mission will study the effects of different shielding materials on UV levels, breakdown of a contained atmosphere due to solar radiation, and detecting and analyzing any residual lunar atmosphere.

- The secondary mission will be to detect lunar seismic activity through use of a seismic probe which will be embedded into the lunar surface and an electronic seismograph located on the lander.

- Use RTG or similar power source for long term, constant power output requirements.

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CHAPTER 1

ORBITAL MECHANICS

CHAPTER 1 - ORBITAL MECHANICS

INTRODUCTION

The orbital mechanics problem for the Lunar Probe involves the launching, transit, and descent phases. Several assumptions must be made in the determination of these mechanics. First, the Earth and the Moon are assumed to be symmetric. Second, the probe is only affected by the Earth's gravity while within the Earth's sphere of influence and the Moon's gravity while in the Moon's sphere of influence. Third, all orbits are Keplerian. Finally, all orbit burns are considered to be instantaneous. The method used to reach the moon was the simplest and required the fewest burns. This method ended with the probe being a "rock" falling directly to the moon.

LAUNCH

The probe will be launched from the Kennedy Space Center on a Delta II. The Delta II will place the probe in Low Earth Orbit (LEO) at an altitude of 5,000 km (orbital parameters of LOE are found in Table 1-1). This is the maximum LOE obtainable by the Delta II and this lowers our DV into our transfer orbit. The probe will then have a PAM-d burn at the appropriate time so the transfer orbit will intercept the Moon's sphere of influence as shown in Figures 1.1 and 1.2.

TRANSFER ORBIT

The transfer orbit is a hyperbolic orbit. It was necessary to use a hyperbola to obtain the type of landing desired. The orbital parameters of the transfer orbit were derived from the velocity desired at 1500 km above the lunar surface. It was determined by the propulsion section that at 1500 km above the lunar surface the probe would be falling straight toward the moon at 2600 m/s. Integrating back to the Moon's sphere of influence, the speed of the probe falling

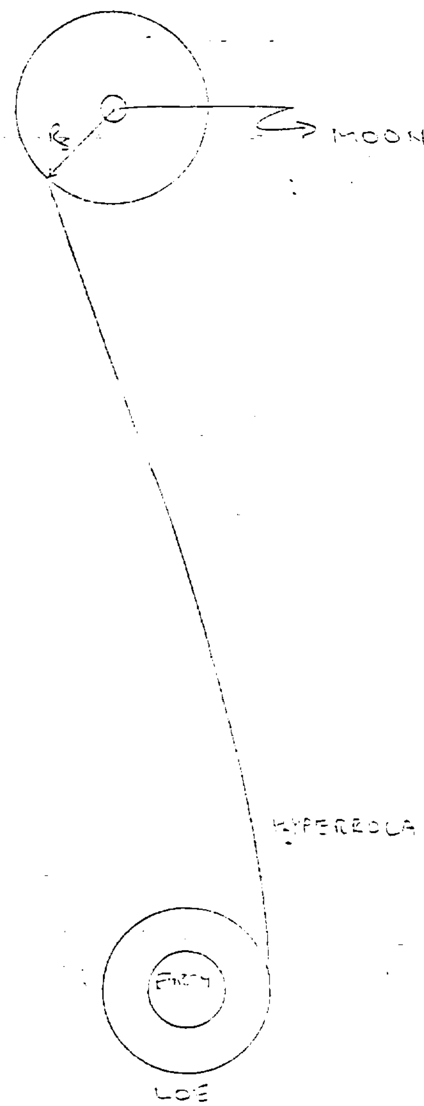

FIGURE 1.1

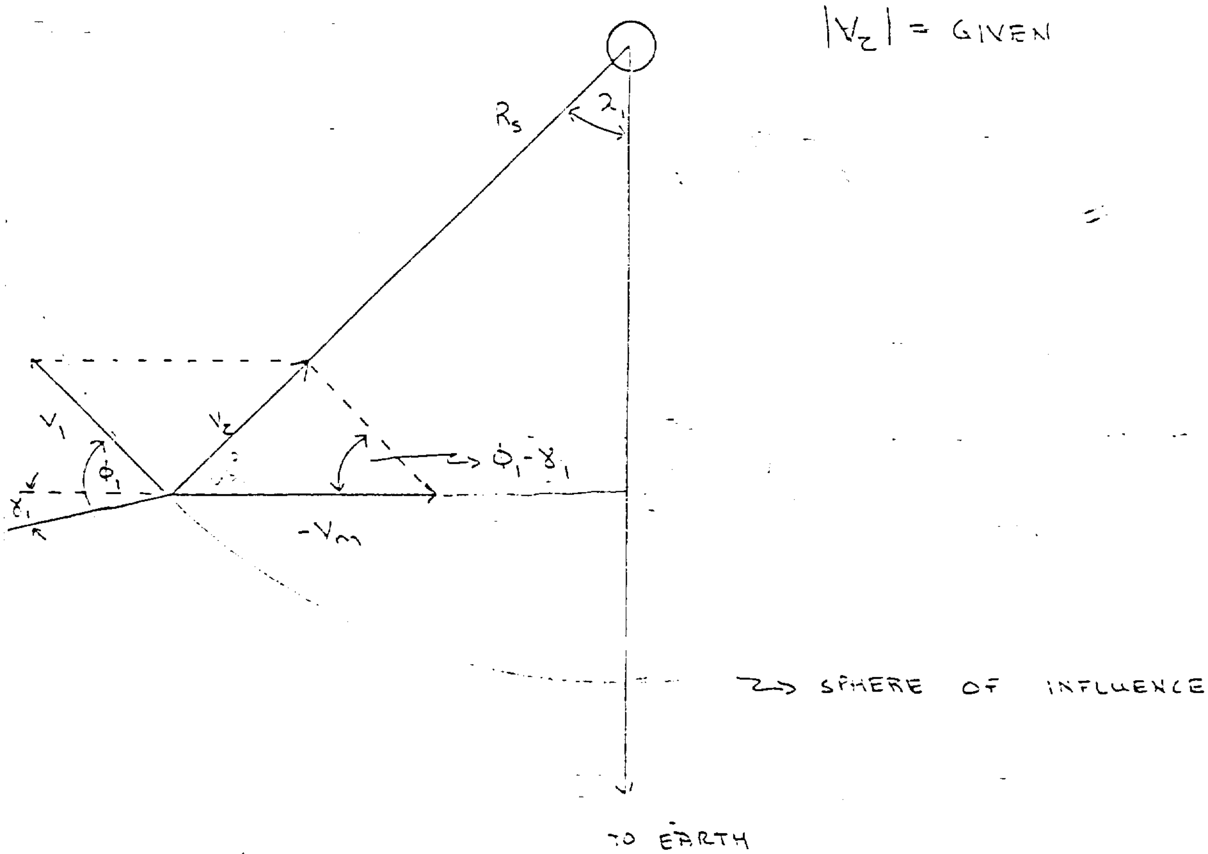

FIGURE 1.2

directly toward the moon was found to be .8451878 km/sec (see Table 1-2). Using vector addition as shown in Fig. 1.2 the velocity vector of the probe (V_1) was determined and the radius of the probe from the earth was determined. Using these two parameters, the orbital parameters of the transfer orbit were determined (see Table 1-1). The result was a hyperbola.

DESCENT

The descent phase is the simplest phase of the orbital mechanics. In this phase the probe enters the Moon's sphere of influence, the probes velocity vector (V_1) and the Moon's velocity vector (V_m) result in a vector pointing straight toward the moon. From here the probe falls like a rock and the decent phase will be controlled as described in the propulsion section of the report (Chapter 3). This type of landing and decent was chosen because of its simplicity. The mission could use one solid rocket to remove most of the velocity from the descending probe and then verniers to control the final portion of landing.

COMPARISON

Calculations were completed for using a hohmann transfer to reach the moon and land, also. The calculations showed the marked difference between using a hyperbola and the hohmann. Although the hohmann had a smaller total DV(see Table 1-3), the hyperbolic approach was chosen due to its simplicity. There were fewer burns and minimal translational motion when using the method discussed above. The chosen method is feasible and has been proven, if the design were to have gone further then much more investigation would be necessary to determine which type of approach to use.

TABLE 1-1 ORBIT DATA

LOE(circle)	Transfer	Moon
h loe= 67345.33	h trans= 2.4451819	mu moon= 0.0123001
a= 11378.266 km	a= hyperbola	h moon= 7.7515557
e= 0	e= 2.3515132	a= 384400 km
p= 11378.266 km	p= 38134.409 km	e= 0.0549005
ra= 11378.266 km	ra= hyperbola	p= 383241.39 km
Va= 5.9187691 km/sec	Va= hyperbola	ra= 405503.75 km
rp= 11378.266 km	rp= 11378.266 km	Va= 0.9638529 km/sec
Vp= 5.9187691 km/sec	Vp= 10.835573 km/sec	rp= 363296.25 km
	DU= 6378.1492 km	Vp= 8.5048535 km/sec
	TU= 806.81187 sec	DU= 1738 km
	DU/TU= 7.9053683 km/sec	TU= 1035 sec
		DU/TU= 1.6792271 km/sec
	Delta v= 4.9168035 km/sec	

TABLE 1-2 FINDING THE TRANSFER ORBIT

Solve for V1	Lambda 1 = 45 deg	V @1500 = 2600 m/s
V1= 0.7654027 km/sec	R1= 396.46051 km/sec	2.6 km/sec
Rs= 66300 km	Gamma 1 = 8.4278324 deg	X1= 1.863061
Rs= 243309.14 km	phi-gam = 51.335266 deg	C1= 0.661914
Distan = 363296.25 km	phi= 59.763098 deg	V2= 0.8451878 km/sec

TABLE 1-3 CALCULATION FOR HOHMANN TRANSFER

v trans for conic		Delt V if use conic
ra= 296996.25 km	h= 147.88574	@ perigee
rp= 11353.106 km	vp= 8.2239614 km/sec	vp trans= 8.2239614 km/sec
e= 0.9263621	va= 0.0004979	vp loe= 5.9187691 km/sec
p= 21870.192 km		delt v= -2.3051923 km/sec

Vel wrt moon	ellpise into moon	Delt V if use conic
V3= 0.76146 km/sec	ra= 38.147296 du m	@ appogee
r= 66300 km	rp= 1 du m	V3= 0.76146 km/sec
V3= 0.76146 du/tu m	e= 0.9489109	Va= 0.0614528 km/sec
r= 66300 du m	p= 1.9489109	delt v= 0.7000072 km/sec
h= 17.298221	h= 1.396034	= 0.7000072 km/sec
p= 299.22844	vp= 1.396034 du/tu m	Total= 3.0051995 km/sec
e= 6.8440277	va= 0.0365959 du/tu m	

TABLE 1.1 EQUATIONS AND METHODS

For a Circular orbit

$r_{circ} = r_p = r_a$ for a circular orbit as does v_p and v_a

$$h = rv\cos(f) \qquad (f = 90°)$$

ϕ

$$V_{circ}=\sqrt{\frac{\cancel{m}}{r_{circ}}} \quad \mu$$

$$2a=r_p+r_p$$

TRANSFER ORBIT

r_p of the transfer orbit = r_{circ}

h of transfer orbit found substituting values of $\overset{\phi}{f}$ (found using trigonometric relations shown in figure 1.2.) r ,and v also derived at sphere of influence into the equation for h shown above. Then v_p of the transfer orbit is found using h found and r_p known and $\cancel{f}\,\phi$ = 90° at the perigee point.

DELTA V

$$DV = V_p\,(\text{transfer}) - V_{circ}\,(\text{LOE})$$

TABLE 1.2 EQUATIONS AND METHODS

First solve for V_2 (fig 1.2) by integration solving for the constant and plug in x= $r_{sphere\ of\ influence}$ to obtain V_2

$$\ddot{X}+\frac{m}{x^2}=0$$

Now using γ=45° (this is the best place by geometry, see fig 1.2) and solving for remaining angles given known distance from the Earth we solve for V_1 using the law of cosines

$$V_1^2 = V_\infty^2 + V_2^2 - 2V_m V_2 \cos(f_1 - g_1)$$

$$\phi_1 - \gamma_1$$

TABLE 1.3 EQUATIONS AND METHODS

LOE remained the same. The ellipse used would have $r_p = r_{circ}$ and $r_a = r_p$ of moon.

$$e=\frac{r_a - r_p}{r_a + r_p}$$

$$p=r_a(1-e)$$

$$h=\sqrt{mp} =$$

Velocity at apogee and perigee in the transfer orbits were found once h was found following the

above steps and then solving for v_p and v_a in the h equation knowing that f=90° at apogee and perigee. Solve for *DV* at perigee by subtracting v_{circ} from v_p.

Now find velocity with respect to the moon using v_{circ} of moon and v_a of transfer orbit (simple subtraction). Everything is now done in reference to the moon. r_a is the sphere of influence of the moon and r_p is the surface of the moon. Solve for the v_a of this ellipse with respect to the moon. Subtract v_a from the velocity with respect to the moon and that is the last *DV* to be determined. Add the two DV's together to get a total for a hohmann transfer.

CHAPTER 2

STRUCTURES AND MECHANISMS

CHAPTER 2 - STRUCTURES AND MECHANISMS

MISSION REQUIREMENTS

1. Be able to withstand a max compressive axial load of 9 g's from the launch phase of the Delta II.
2. Be able to radiate vast amounts of waste heat through the use of dissipation panels.
3. Survive a controlled crash on the lunar surface (approx. 10 g's) so that sensitive payload modules will continue to function.
4. Expose eight gas filled modules to solar radiation and record data.
5. Have a ten year mission life.

LANDER DESIGN

The design of the lunar lander was driven by mission requirements from the beginning. The total weight of the mission required systems came to 112.2 kg. With a star 30E braking motor to slow down the satellite, the Delta II launch platform was selected. The Delta II could lift about 1300 Kg to the moon. Figure 2-1 shows the shroud area that can be occupied by a satellite carried aboard the Delta II. The surface area of the satellite was driven by the size of the panels to dissipate the waste heat from the RTG. The final size calculated will fit into the shroud area. Figure 2-2 shows the top view of the satellite and the folding panels used for the heat dissipation.

The placement of most of the internal components was carefully considered as well. Most of the heavy items like the RTG and most of the Comms/Data Storage equipment was located within the central thrust tube. This served to keep the moments of inertia low in the x-y plane (see Table 2-1) and so reduce weight for attitude control motors. Further, the strongest part of the lunar lander is this thrust tube, and if most of the key components are located within this

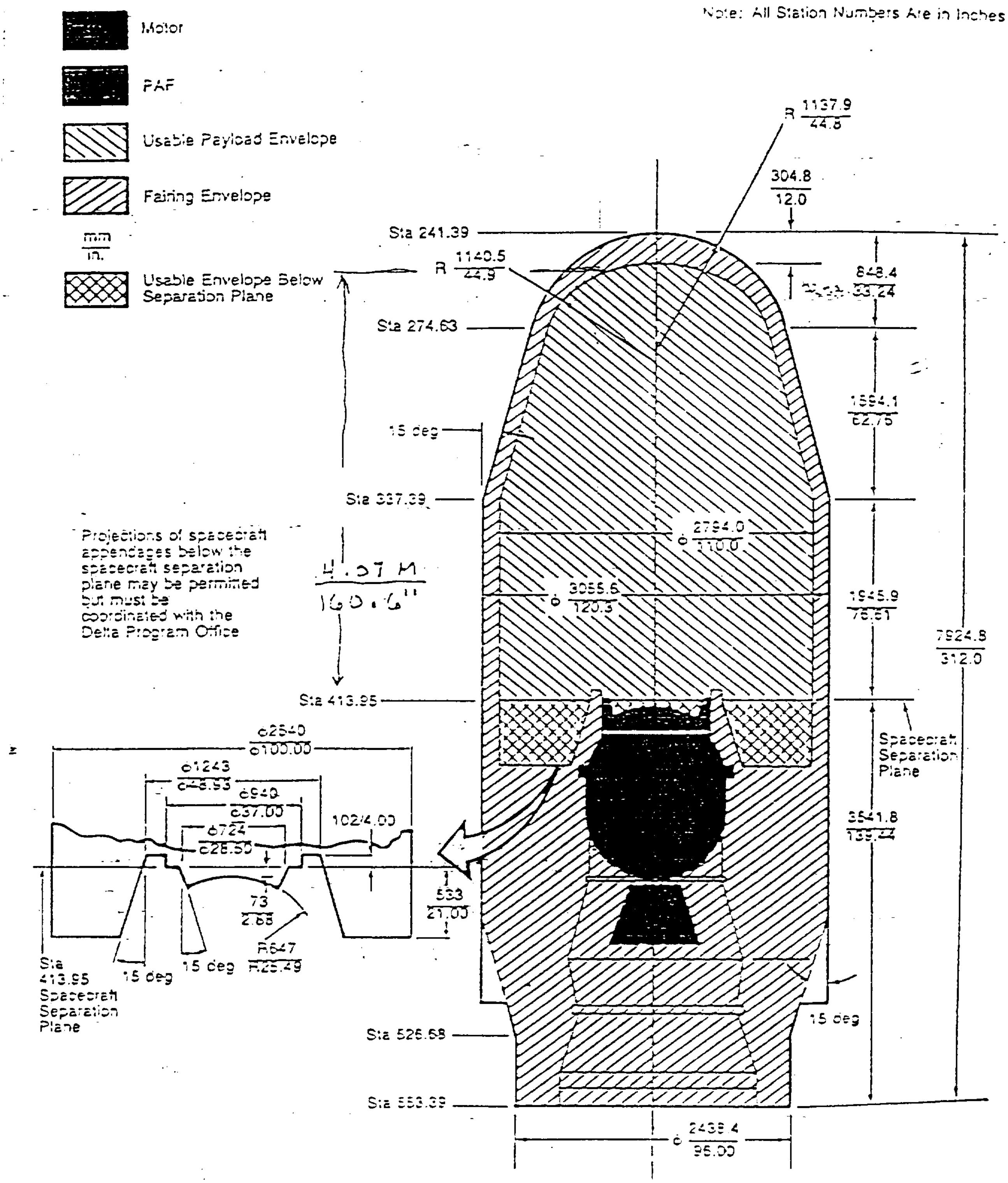
Note: All Station Numbers Are in Inches
Motor
PAF
Usable Payload Envelope
Fairing Envelope
mm/in.
Usable Envelope Below Separation Plane
R 1137.9/44.8
304.8/12.0
Sta 241.39
R 1140.5/44.9
848.4/33.24
Sta 274.63
1594.1/62.75
15 deg
Sta 337.39
ø 2794.0/110.0
ø 3055.6/120.3
1945.9/76.61
7924.8/312.0
Projections of spacecraft appendages below the spacecraft separation plane may be permitted but must be coordinated with the Delta Program Office
4.07 M
160.6"
Sta 413.95
Spacecraft Separation Plane
ø2540/ø100.00
ø1243/ø48.93
ø940/ø37.00
ø724/ø28.50
102/4.00
3541.8/139.44
533/21.00
73/2.88
R647/R25.49
Sta 413.95 Spacecraft Separation Plane
15 deg
15 deg
15 deg
Sta 526.68
Sta 553.39
ø 2438.4/96.00

FIGURE 2-1

TOP VIEW

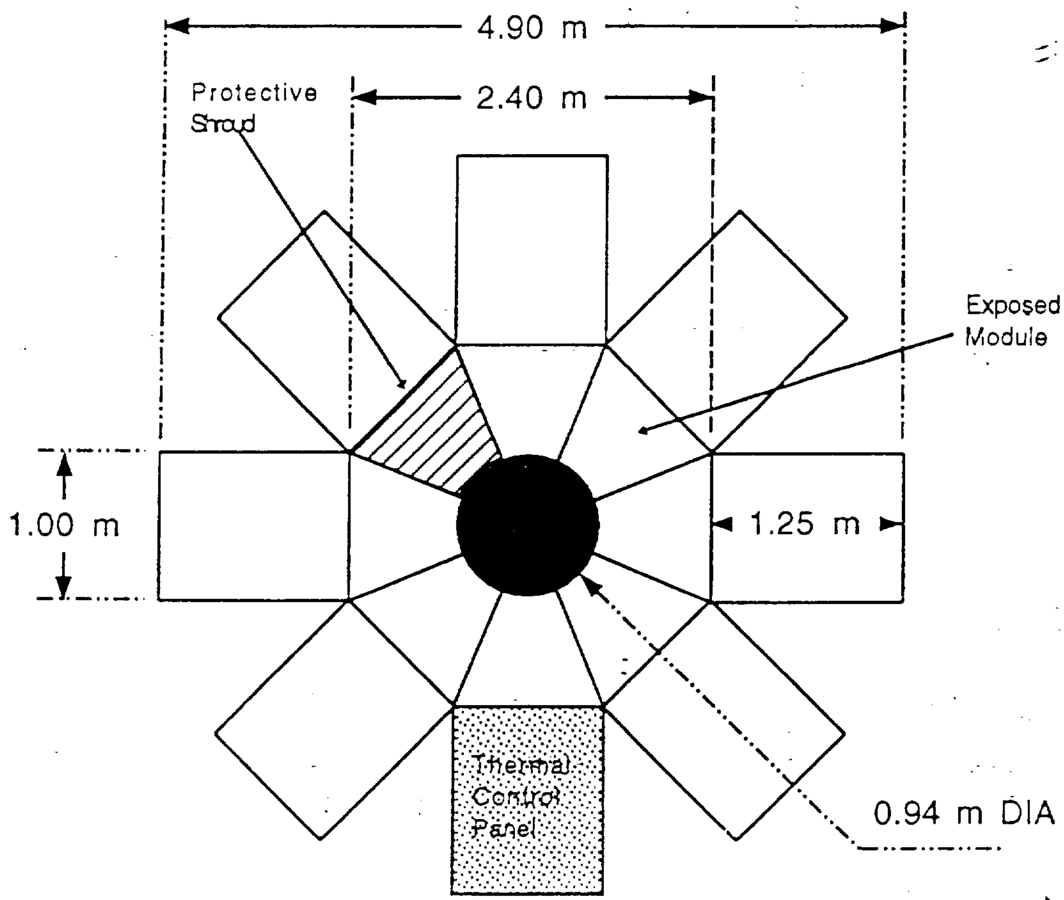

FIGURE 2-2

TABLE 2-1 CALCULATION OF MOMENTS OF INERTIA

Component	Mass	Ix	Iy	Iz	x-pos	y-pos	z-pos	Ixx	Iyy	Izz
RTG	55.9				0.00	0.00	-0.55	0.0000	0.0000	17.1442
Comm. Dish	3.5	0.1852	0.1852	0.3703	0.00	0.00	0.93	0.1852	0.1852	3.3975
Elec. motor	1.2				0.00	0.00	0.85	0.0000	0.0000	0.8670
Mass spec.	3.9				0.00	0.31	0.31	0.0000	0.3628	0.3845
Seismic equip.	10.0				-0.46	0.00	-0.63	2.1160	0.0000	3.9690
Gyro/Accel.	0.5				0.00	0.31	0.00	0.0000	0.0493	0.0000
Verniers	6.9				0.46	0.46	-0.63	1.4600	1.4600	2.7386
Star Tracker	1.0				-1.20	0.00	0.00	1.4400	0.0000	0.0000
Laser-1	1.0				1.04	1.04	-0.82	1.0816	1.0816	0.6724
Laser-2	1.0				-1.04	1.04	-0.82	1.0816	1.0816	0.6724
Laser-3	1.0				1.04	-1.04	-0.82	1.0816	1.0816	0.6724
Comms. controller	1.5				0.00	0.00	0.32	0.0000	0.0000	0.1536
Data recorder	3.4				0.00	0.00	0.26	0.0000	0.0000	0.2298
Multiplexers/filters	2.0				0.00	0.00	0.22	0.0000	0.0000	0.0968
Transceiver	13.8				0.00	0.00	0.16	0.0000	0.0000	0.3533
Radiation detectors	4.8				0.60	0.60	0.63	1.7280	1.7280	1.9051
Voltage regulator	0.8				0.00	0.00	0.00	0.0000	0.0000	0.0000
total structure	4.203	3.5684	3.5684	6.0246	0.00	0.00	0.00	3.5684	3.5684	6.0246
thrust tube	5.42	1.2770	1.2770	1.1198	0.00	0.00	0.00	1.2770	1.2770	1.1198
fuel ring	0.7351	0.0989	0.0989	0.1978	0.00	0.00	0.00	0.0989	0.0989	0.1978
legs (3)	3.73	0.1463	0.1463	0.1463	0.00	0.00	-0.82	0.1463	0.1463	2.6543
Star-30 spacer ring	1.14	0.1982	0.1982	0.2355	0.00	0.00	-1.04	0.1982	0.1982	1.4685
aluminum crash block	1.28	0.0749	0.0749	0.1354	0.00	0.00	-0.76	0.0749	0.0749	0.8748
Star-30 Motor	667	49.3313	49.3313	49.3313	0.00	0.00	-1.04	49.3313	49.3313	770.7585
Fuel spheres	200	0.2000	0.2000	0.2000	0.52	0.52	0.06	53.3893	53.3893	0.8272
								Ixx tot	Iyy tot	Izz tot
								118.2582	115.1143	817.1822

BOTTOM VIEW

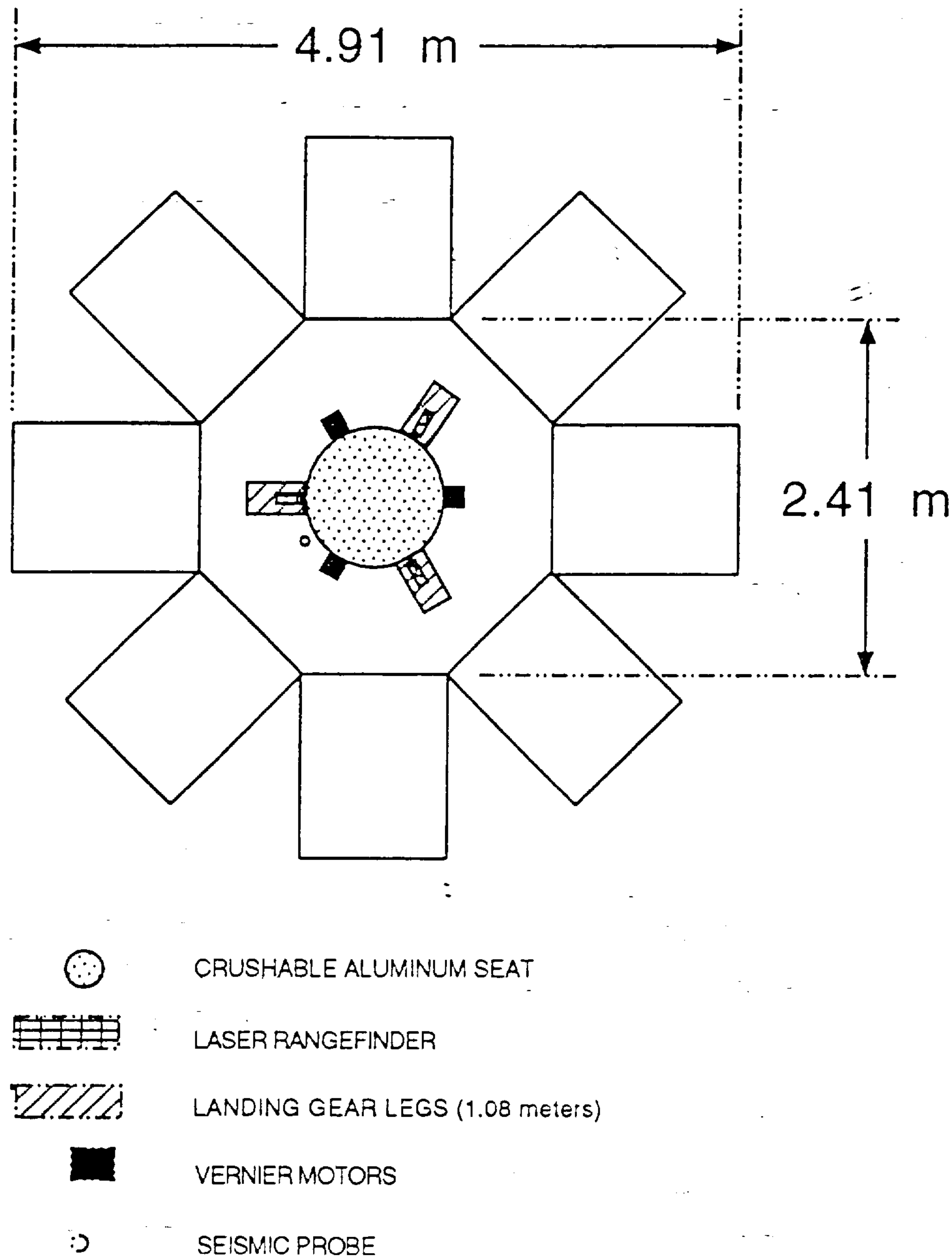

FIGURE 2-3

SIDE VIEW

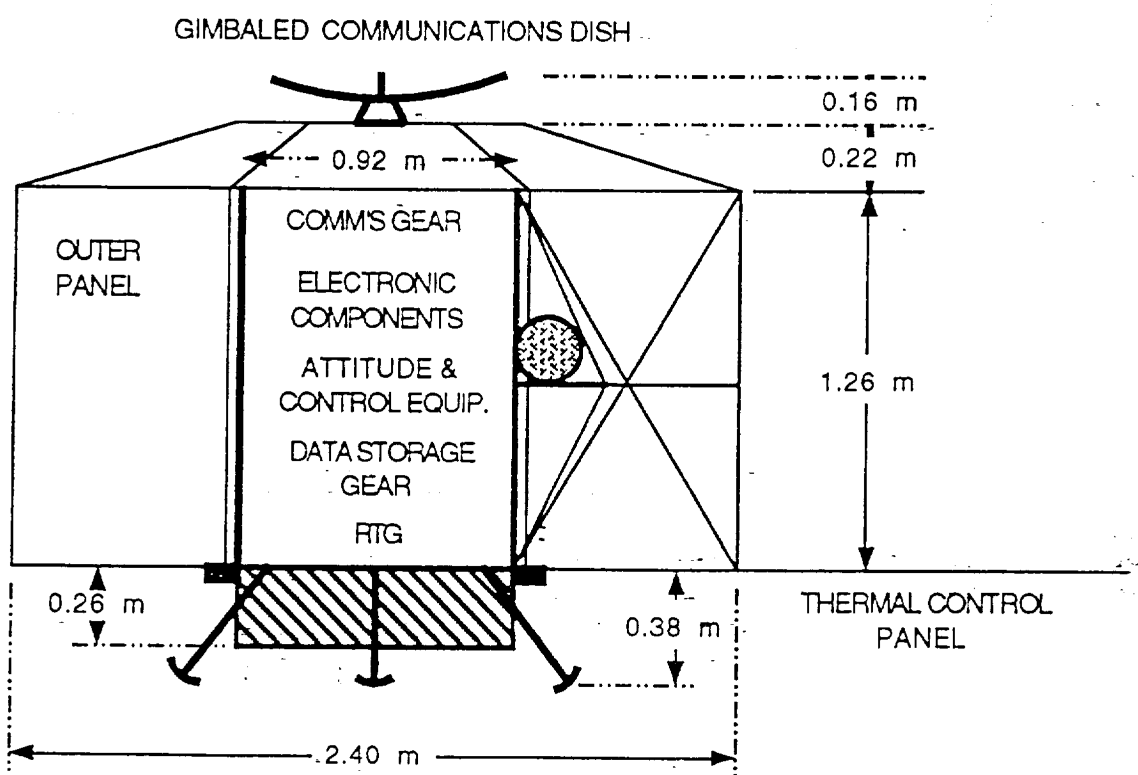

FIGURE 2-4

cylinder, they are protected, and the bending moments caused by the items is reduced to a minimum.

Figure 2-5 shows a summary of component weights and the obvious benefits of reducing the need to support such items along any members.

The lunar lander's materials were chosen both to survive the expected ten year mission life and to keep the weight down to a minimum. Figure 2-6 shows a table of individual weights of panels, shrouds, stringers and other structural components. Most of the non load bearing members are made of honeycomb aluminum that is 1/4 inch thick. This material weighs only .725 Kg per cubic foot. The thrust ring and other load bearing components are made of 1/2 inch thick aluminum that only weighs 1.41 Kg per cubic foot. The lander legs and stringer supports were all fashioned from 6061 T6 aluminum that weighs 76.81 Kg per cubic foot. Aluminum was chosen as the primary building material for its excellent strength and light weight. It has excellent heat dissipation properties, and will have very little trouble lasting the mission design life of ten years.

Finally there is the question foremost on the mind of any designer of space vehicles. How much does the system weigh? Figure 2-7 shows the bottom line weight values. It shows the expected lift capacity of the Delta II, the designed weight of the satellite, and the difference. By coming in over 15 percent under weight, room has been created for any unforeseen design or requirement changes.

WEIGHTS OF COMPONENTS AND SUBSYSTEMS

ITEM	INDIV. WGT	TOTAL WGT	SATELLITE WGT.
RTG	55.9 Kg	55.9 Kg	55.9 Kg
Comm. Dish	3.5 Kg	3.5 Kg	59.4 Kg
Elec. Motor	1.2 Kg	1.2 Kg	60.6 Kg
Mass Spectr.	3.9 Kg	3.9 Kg	64.5 Kg
Seismic Equ.	10.0 Kg	10.0 Kg	74.5 Kg
Gyro./Acell	.5 Kg	.5 Kg	75.0 Kg
Verniers	2.3 Kg	6.9 Kg	81.9 Kg
Star Tracker	1.0 Kg	1.0 Kg	82.9 Kg
Lasers	1.0 Kg	3.0 Kg	85.9 Kg
Comms. Controller	1.5 Kg	1.5 Kg	87.4 Kg
Data Recorder	3.4 Kg	3.4 Kg	90.8 Kg
Multiplexer/ Filters	2.0 Kg	2.0 Kg	92.8 Kg
Transceiver	6.9 Kg	13.8 Kg	106.6 Kg
Radiation Detectors	.6 Kg	4.8 Kg	111.4 Kg
Voltage Regulator	.8 Kg	.8 Kg	112.2 Kg
Lander Structure	123.4 Kg	123.4 Kg	235.6 Kg
Total Payload/Structure weight ====================			235.6 Kg
Propulsion Systems:			
Star 30E	667.0 Kg	667.0 Kg	667.0 Kg
Fuel/ Subsystems	200.0 Kg	200.0 Kg	867.0 Kg

FIGURE 2-5

SUMMARY OF LANDER STRUCTURE WEIGHTS

ITEM	WEIGHT	TOTAL WEIGHT
Top Panel	.7359 Kg	.7359 Kg
Bottom Panel	.7359 Kg	1.471 Kg
Shroud	.7359 Kg	2.208 Kg
Side Panels	.2494 Kg	4.203 Kg
Structural Stringers	3.08 Kg	78.12 Kg
Thrust Tube	5.42 Kg	83.54 Kg
Fuel Ring	.7351 Kg	84.28 Kg
Fuel Ring Stringers	1.57 Kg	109.41 Kg
Landing Legs	1.91 Kg	115.14 Kg
Comm Antennae Spacer Ring	.7810 Kg	115.92 Kg
Star 30 Spacer Ring	1.14 Kg	117.07 Kg
Aluminum Crash Block	1.28 Kg	118.34 Kg
Asstd. Hardware	5.0 Kg	123.35 Kg

Total Lander Weight ============================ 123.35 Kg

All top, side, bottom and shroud panels are constructed of 1/4" thick honeycomb aluminum weighing .725 Kg per cubic foot. The thrust ring and fuel support ring are made of 1/2" honeycomb aluminum weighing 1.41 Kg per cubic foot. The structural and fuel support stringers are made of 6061 T-6 aluminum weighing 76.81 Kg per cubic foot. The landing legs are constructed of the same 6061 T-6 aluminum.

FIGURE 2-6

Total Fuel/Fuel Subsystems weight ================ 867.0 Kg

Total of Payload/Structure/Fuel/Subsystems ======= 1102.5 Kg

Expected lift capacity of Delta II
w/ PAM-DII to moon =============================== 1300.0 Kg

% Difference between design weight and
max lift capability of Delta II ================= 15.2% less

FIGURE 2-7

CHAPTER 3

PROPULSION

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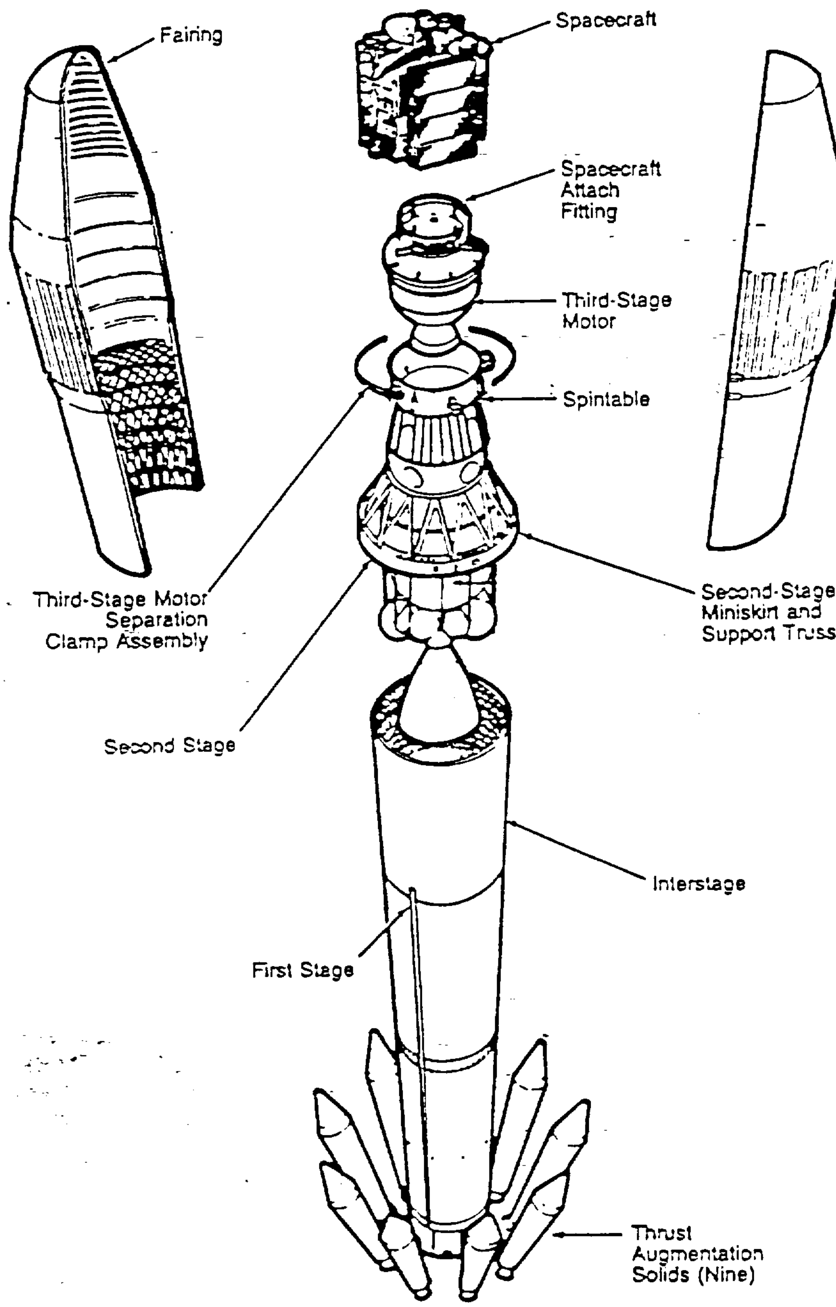


Figure 3-1 Typical Delta II Three-Stage Separation

attach a long slender rod to the end of the lander that will act as a sensor for impact. When the lander impacts the surface, it drives the spacecraft into the surface and triggers and ejection of the spherical lander. The lander then bounces along the surface at which time several pedals open up and ensure the proper orientation. The advantage of the system is that it is extremely simple and inexpensive. It is, however, only intended for small payloads and is untried by the West. Additionally, the bounce procedure would cause a random final orientation of the lander.

Utilized by the Surveyor lunar lander series, the two-level system combines two propulsion systems into three phases. The first phase is a large, main retro burn intended to remove the majority of kinetic energy from the lander. The next phase employs vernier motors to "tilt" the thrust axis in order to establish zero lateral velocity. The final phase then powers up the vernier motors to full thrust and takes the lander to a near hover at a predetermined altitude at which time the lander free-falls to the surface. The two-level system has the advantage of combining a "brute" system with a "finesse" system and has been proven. It utilizes reliable components and has the added option of jettisoning the main retro prior to landing. The problem with the two-level approach is that it is a more complex technique involving several phases and has a moderate cost.

The lunar lander mission being designed needs to have a controlled landing orientation and location in order to assist in the seismic and atmospheric measurements, and needs to prevent any type of post landing interferences with payload instruments. Additionally, since the landing procedure is critical to mission success, it needs to maximize reliability. The two-level approach was therefore chosen because it met each of the design needs with minimal drawbacks.

MAIN RETRO

The next step was to select the type of system to be used for the Main Retro. The options considered were liquid, solid, and advanced technologies.

The advantages of liquid systems are their high performance and the ability to either be throttleable or to be turned off and on. Proven liquid systems are also available, however they are complex and expensive. The increase in complexity has an additional drawback in that it causes an increase in size and mass of the system.

Solid rocket systems have the advantage of being fairly simple and have low volume. They also have a low cost, moderate performance, and proven systems are available. The only major drawbacks to solid rocket motors is that they cannot vary thrust except through modification of the burn core area, and they cannot be turned off.

Several advanced technology propulsion systems, such as nuclear and hybrid systems offer the potential for high performance and efficiency. The systems are unproven and costly though, and most of the proposed systems are very complex.

The Lunar Lander Main Retro needed to maximize reliability and be jettisonable in order to prevent interference with payload instruments. It only needed to have a constant thrust and a single burn, and needed to be compatible with the Delta II nacelle and Lunar Lander structure. The liquid motor would be expensive, large, and difficult to jettison. The advanced systems were unproven and also costly. The solid motor met all of the criteria, and at minimum cost and was therefore selected as the motor type for the main retro.

VERNIERS

Several propellant options were considered for the vernier motors. They included cold gas, bipropellant, and monopropellant.

Cold gas systems are extremely simple, reliable, and can be obtained at low cost. They have very low performance though, and are also extremely heavy for their given performance. Bipropellant systems are very high in performance and there are proven systems available. Their propellants however, are often toxic, they systems are complicated, and have a moderate to high cost. Additionally, some bipropellant systems increase complexity through the numerous storage considerations such as boil-off calculations, and insulation.

Monopropellant propulsion systems are simple, reliable, and available at a low cost. The disadvantage to monopropellants is that they have lower performance and are heavier that bipropellants.

The lunar lander vernier system needed to maintain structural compatibility with the lander sensors, minimize risks to mission failure and provide a range of thrust values acceptable for attitude control and for landing. The monopropellant was therefore selected due to the simplicity, reliability, and low cost of the system.

TWO-LEVEL SYSTEM LANDING REQUIREMENTS

The following landing procedure requirements were developed for the Lunar Lander:

- Landing procedure starting point at 1500Km and 2600m/s. This point is based on historical data (Surveyor) and serves as a starting point for the main retro burn. The altitude is roughly twice that of surveyor to ensure enough time to complete the second phase of decent. The velocity also serves as the landing delta V required.

- Minimum sustained thrust of 40% hover. This is the minimum sustained thrust used for Apollo and prevents the lander from obtaining large velocity gains during minimum thrust periods.

- At least 95% reduction in velocity from Main Retro Burn. This requirement is based on historical data (Surveyor) and is intended to ensure the majority of the velocity change is made by the Main Retro.

- Less than five vernier motors. This requirement is simply to minimize complexity and avoid thruster packs.

- Compatible to structural specifications. These specifications include fitting sizes, maximum stresses, and thrust axis vectors.

- Proven system, in order to maximize reliability.

- Approximately 200 kg for payload and 1100 Kg for main retro system and vernier propellant.

COMPONENT SELECTION

The main retro was selected first in order to meet design criteria of 95% reduction of velocity and a proven system. The STAR motor (for a list of possible motors see Figure 3-2) provided several variants which had all been tried and proven. The variant providing the maximum thrust, with a loaded mass under 1000kg is the STAR 30E (mass = 667kg). The change in velocity under full loading conditions was calculated using Tsiolokovsky's equation and was found to produce a 96.4% reduction in velocity.

Tsiolokovsky's Eqn:

$$\text{delta } V = g * Isp * \ln[(Mo)/(Mo-Mp)] \quad (3\text{-}1)$$

Motor	Total Impulse (N·s)	Loaded Weight (kg)	Propellant Mass Fraction	Avg. Thrust (lb_f)	Avg. Thrust (N)	Max. Thrust (N)	Effective I_{sp} (s)	Status
IUS SRM-1 (ORBUS-21)	2.81×10^7	10,374	0.94	44,610	198,435	260,488	295.5	Flown
LEASAT PKM	9.26×10^6	3,658	0.91	35,375	157,356	193,200	285.4	Flown
STAR 48A	6.78×10^6	2,559	0.95	17,900	79,623	100,085	283.9	Flown
STAR 48B(S)	5.67×10^6	2,135	0.95	14,845	66,034	70,504	286.2	Qualified
STAR 48B(L)	5.79×10^6	2,141	0.95	15,160	67,435	72,017	292.2	Qualified
STAR 62	7.12×10^6	2,459					293.5	In development
STAR 75	2.13×10^7	8,066	0.93	44,608	198,426	242,846	288.0	In development
IUS SRM-2 (ORBUS-6)	8.11×10^6	2,995	0.91	18,020	80,157	111,072	303.8	Flown
STAR 13B	1.16×10^5	47	0.88	1,577	7,015	9,608	285.7	Flown
STAR 30BP	1.46×10^6	543	0.94	5,960	26,511	32,027	292.0	Flown
STAR 30C	1.65×10^6	626	0.95	7,140	31,760	37,031	284.6	Flown
STAR 30E	1.78×10^6	667	0.94	7,910	35,185	40,990	289.2	Flown
STAR 37F	3.02×10^6	1,149	0.94	9,911	44,086	49,153	291.0	Flown

Figure 3-2 Solid Rocket Motor Specifications

In order to maximize stability, while conforming to design criteria of under five vernier motors, a number of three motors was chosen.

The vernier motors were selected to meet the design criteria of minimum thrust at 40% hover thrust. Research of proven mono H motors indicated that several systems had been widely used successfully. Rocket Research Company's MR-104 445N motors provide the necessary thrust and a relatively high Isp (239s). Calculations indicate that using three of these motors would require approximately 85kg (84.47) of propellant to complete the remaining delta V. Additional propellant would be required however, for the lateral burn, the midcourse correction, and for attitude control.

The MR-104 motors have been used previously on Magellan and Voyager missions for attitude control. The have a sustained variable thrust of 205-572N. Additionally, they have a minimum pulse duration of 0.022 seconds. Equation 3-2 was then used with moment of inertia calculations to indicate a minimum angular velocity correction of approximately .025 rad/s in two

of the axes and .0025 rad/s in the third axis.

w = [(delta T)*(F)*(D)] / I (3-2)

delta T = time of burn

w = angular velocity

F = thrust

D = distance to axis

I = moment of inertia

CHAPTER 4

ATTITUDE DYNAMICS AND CONTROLS

Chapter 4 - Attitude Control System

Requirements

This area of mission design has several requirements that must be met by a design that can operate autonomously. The first major function of this system is to verify that the spacecraft has achieved the proper parking orbit around earth. Once this is accomplished the attitude control system must ensure the spacecraft has the proper alignment at the right position in orbit to ensure a proper perigee kick to begin the earth-moon transit. During the transit to the moon, the attitude determination system will perform the navigational portion of its mission by doing midcourse position checks. From these readings it must carry out any necessary course corrections. Once in the vicinity of the moon, the spacecraft must be oriented so that the main retro engine is pointed directly opposite the velocity vector; this will consist of fine adjustments, as the retro engine will be mounted on the "front" end.

The most critical segment of the lunar mission is getting the payload safely down to the lunar surface at under 10 g's. To perform this feat, the attitude determination and control system must be able to read lateral and vertical ranges and rates of descent. It must also be able to correct for any errors with the above readings. Gravity torques are less than 0.001 N•m at a maximum. Finally, a whole-flight concern for the ADCS is to have the capability to correct for any contingencies that may arise.

Spacecraft Control Type

In the final phase of the lunar mission, the spacecraft must have the ability to rotate in any direction and correct for drift and descent rates. To reduce multiplicity of ADCS hardware, it

was decided that this three-axis stabilization method was to be used throughout the mission. Due to the complexity of the maneuvering requirements, it would obviously be simpler to control the lunar descent phase from earth. The problem with this solution is that the lag time that results from attitude data transmission and correction transmission was too long to compensate for errors encountered once the lander got close to the lunar surface. This condition creates the need to have the descent mode of attitude control handled by an autonomous system on board the lander.

Sensor Selection

Whichever attitude sensors were decided upon for use would have to be used for each phase of the mission. This would help to keep the weight at a minimum so the lander could be sent aloft in the less expensive launch vehicle that was chosen. In the first phase of the mission, the spacecraft is in a low-earth parking orbit. In this regime magnetometers provide moderate accuracy, but their performance degrades as distance from earth increases. Horizon sensor accuracies are generally better, but suffer from the same problem. Using a sun sensor here might improve accuracy by a factor of ten, but the perigee kick might need to occur at a period in which the spacecraft is in eclipse, eliminating the availability of data. Star sensors and inertial measurement units would also yield high accuracies, plus they could be used during other phases of the mission.

The next phase of the mission involves the transit to the moon. During this phase, the sensors relying on earth for measurements would become increasingly unreliable as the distance to earth increased. However, the sun sensor would maintain slightly higher reliability than what it exhibited in LEO due to the absence of cyclic eclipse periods. The star sensor and inertial measurement options would continue to display approximately the same error as before, making

these options the best suited for the mission. The only problem here is that there will be a certain drift error associated with the inertial measurement units. To solve this drift problem, the data from the star sensor can be used to update the inertial units and keep them set to a single, constant reference system. It was finally decided that a ring laser gyro (see Figure 4-1) would be used to capitalize on the accuracy, light weight, and reliability of strap down technology. A star tracker (see Figure 4-2) was also decided upon for its reliability and light weight.

For navigation, all terrestrially-based methods, such as GPS or tracking radars or satellites, were deemed unsatisfactory due to the differing flight regimes within the mission. The space sextant could operate beyond LEO; however, using it would require a large weight and power allotment. Also, it is not currently being marketed for use. The final means of navigation examined was the Microcosm Autonomous Navigation System (MANS), which proved to be ideal. It added little or no weight or power requirements, as it could operate using existing attitude sensing hardware, which had already been selected. MANS was also designed for lunar and planetary orbits as well as low-earth and geosynchronous orbits.

The final phase of the mission is unlike the other phases. The attitude detection methods must be completely different due to the simple fact that the payload is landing on the lunar surface. In this regime, rapid detections and calculations must be made to determine how fast the lander is approaching the surface. In addition, the ADCS must solve for and execute corrective measures to ensure a proper landing as defined by the requirements. To meet these guidelines, some sort of active sensing technique must be used so that a return signal can be compared to an expected value. From this difference the system can quickly and accurately determine the desired ranges and range rates.

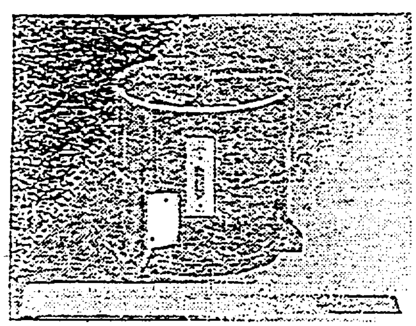
For this phase, one technique that was looked at was Doppler radar technology. This, however, would create prohibitive power requirements if it were to be used over 100 km. An alternative to the radar system was to use a lighter-weight, farther-reaching laser range finding

Inertial Measurement Units (IMU)

- 1st Orbital Flight Qualification Of Lightweight Inertial Measurement Units
  - Spacecraft Incorporates Ring Laser Gyro (RLG) & Interferometric Fiber Optic Gyro (IFOG)
  - Incorporate Lightweight Packaging & Applications Specific Integrated Circuits (ASIC) For 2:1 Weight Reduction Over Existing IMUs
  - Radiation-Hardened Designs To 10 KRads(Si)

Parameter	RLG	IFOG
Bias - Long Term (deg/hr)	1.0°	1.0°
Scale (ppm)	50	50
Random Walk (deg/root-hr)	.125	.005
Size (cm)	7.7 Dia x 12.7	8.9 Dia x 7.9
Volume (cm^3)	2310	1950
Weight (g)	500	650
Power (W)	11	10

Ring Laser Gyro



Interferometric Fiber Optic Gyro

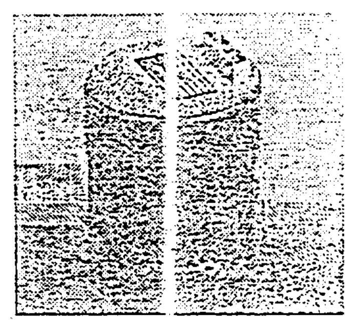


FIGURE 4-1

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Star Tracker Camera

- Miniaturized, Medium Accuracy (100 - 300 mrad) Optical Sensor
  - Fast, Robust 3-Axis Attitude Determination With Only One Starfield Image
  - Star Tracker Wide-Field-Of-View (WFOV) Camera Minimizes Star Catalog By Using Brightest Stars
- Current Star Trackers Are 3x -10x Heavier, 2x Power Consumption & 2x - 5x Expensive
- Fast & Accurate Attitude Sensor For DoD Missions
  - Spacecraft Processor Performs Algorithm Calculations To Determine 3-Axis Attitude
  - Attitude Determination Is 10x - 100 x (< 1 Second) Faster Than Current Sensors
  - Designed For Inexpensive Manufacture & Calibration
  - Common Control & Data Bus Architecture For Ease Of Integration & Test

Miniaturized Star Tracker Camera Configuration	
Mass (grams)	370
Size (cm)	12 x 12 x 14
Electrical Power (Watts)	7
Field Of View (degrees)	29 x 43
Pixel Format	384 x 576

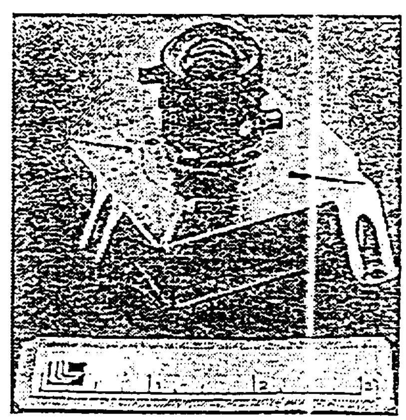

FIGURE 4-2

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system. The problem here is that this arrangement would not directly provide range rate data. To get around this, a range calculation will be made four times every second. The rate would be calculated by dividing the range differences by the time differential. As the lander gets closer to the lunar surface, the range sampling rate will be increased to provide for more rapid and accurate descent data so more precise adjustments can be made.

Each of three laser range finders (see Figure 4-3) will be placed on a landing leg on the bottom of the lander structure to provide an unobstructed field of view. To provide for lateral motion detection, the lasers will be angled 15° outward from vertical. This angling technique will also serve another purpose in ensuring sensor contact with the lunar surface in the event a maneuver causes the other laser(s) to swing above the lunar horizon. Using 15° will allow the sensory cone to intersect the lunar limb when the lander is at an altitude of 4950 km. This will allow for final spacecraft-lunar alignment well before main retro burn.

Hardware Selection

For three-axis stabilization, a spacecraft can point itself using thrusters, momentum wheels, control moment gyros, and magnetic torquers. Since the lunar lander will be making a transit away from the earth, using magnetic torques becomes ineffective at great ranges. Using a set of control moment gyros would drive the weight of the ADCS hardware up compared to using existing hardware. Using momentum wheels would be satisfactory until the spacecraft reached the moon. Once in descent, the increased control requirements might drive the wheels into saturation making them useless until momentum dumping could be accomplished, but lunar descent is not the ideal time for desaturation. This eliminates all types of actuation but thrusters, and since there are already thrusters for the mission, fuel can be added so the vernier motors can carry out attitude control.

Laser Transmitter

- Flight Qualification of Compact, Short-Pulsed Laser & Beam Expander
- Previous Lasers 10x Larger & Heavier
- Ranges Small Objects (Tens Of Kilometers) & Large Objects (Hundreds Of Kilometers)
- Module Laser Transmitter & Power Supply Design Eases Placement & Thermal Control Issues

Mass (grams)	1000
Size (cm) Laser XMTR Power Supply	 4 x 10 x 22 2.5 X 10 x 16
Power (Watts)	5
Laser Type	Diode-Pumped Nd-YAG
Conversion Efficiency	3.5%
Wavelength (microns)	1.064
Pulse Energy (Joule)	0.18
Pulse Length (nsec)	10
Repetition Rate (Hz) Continuous Burst	 1 8

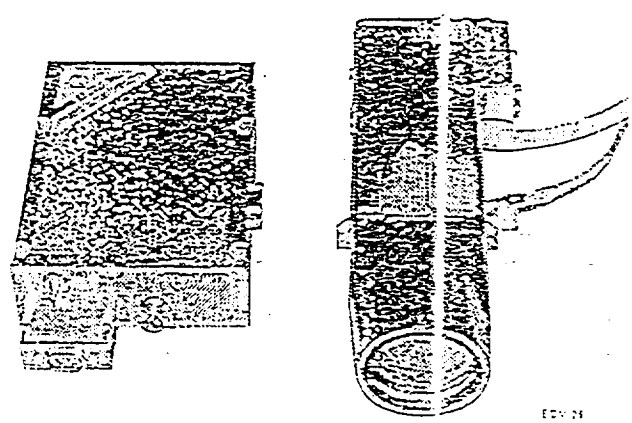


FIGURE 4-3

Attitude Control and Dynamics Summary

Component	Power (W)	Voltage (V)	Mass (kg)
Laser Gyro & Accel.	10	15	0.5
Star Tracker	4.5	25	1.0
Laser Range finders (3)	15	25	3.0

- The ring laser gyro and accelerometer package will be able to detect rotational and translational accelerations, while the supporting software can integrate to determine velocity and position during earth orbit and transfer.

- The star tracker will provide periodic updates of the accelerometer suite to ensure accurate readings with reference to a fixed coordinate system.

- Navigation during the Earth-Moon transit will be handled by the Microcosm® Autonomous Navigation System as an add-on to the existing attitude determination equipment. This was chosen based on its autonomy and ability to combine with current equipment. Added positive features are that it adds very little to mass and power budget, has an accuracy of up to 400 meters, and can be used for ranges up to the lunar and planetary scale.

- The laser range finders will be spaced at equal angles on the legs around the base of the landing craft.. They will be aimed 15° outward from vertical to maintain contact by at least one beam in the event a maneuver swings the other beam(s) above the lunar horizon. This 15° allows

all three beams to intersect the lunar surface when the spacecraft is just under 5000 km above the deck, well before the main retro burn.

- The range rate will be calculated by taking four altitude samples per second and applying the time differential. As the craft approaches the deck the sample rate can be increased to accommodate the need for greater accuracy. Once on the lunar surface, power can be transferred from the attitude control to equipment to other systems.

CHAPTER 5

POWER

CHAPTER 5 - POWER

INTRODUCTION

The purpose of this chapter is to provide information about the power requirements of the lunar lander mission, the choices of power subsystems available, the implementation of a power subsystem (specifically the Radioisotope Thermoelectric Generator, or RTG), and the problems encountered in its design. Overall, this chapter will step through the design of the power subsystem for the lunar lander mission.

REQUIREMENTS

The design of the power subsystem of the lunar lander revolved around three basic

1. Supply 150 Watts (BOL) power to the subsystems and payload, as required, for a 10-year mission with an End-of-Life Power of approximately 97 Watts.
2. A compact, low weight power source to fit within the relatively small-mass lunar lander.
3. Supply constant power to carry out the mission objectives of:
   a. Testing seismic activity on the moon
   b. Testing the effects of radiation on different materials within enclosed structures.

To meet these requirements, a power system had to be designed. This design was chosen from numerous capable, yet proven systems.

Before exploring the various possible subsystems for use in this mission, the power and voltage bus for the entire lander must be examined (see Table 5-1).

TABLE 5-1 POWER AND VOLTAGE BUS FOR LUNAR LANDER		
Subsystem	Power (W)	Voltage (V)
Attitude/Control		
Laser gyro & Accelerometers	10	15
Laser Range Finder (3 @ 5W/each)	15	25
Star Tracker	4.5	25
Communications		
Transmitter	25	25
Receiver	7	25
Solid State Data Recorder	15	25
Spacecraft Controller	10	25
Thermal Control	10	25
Propulsion	0	28
Payload		
Seismograph	3.24	21
Mass Spectrometer	4.5	25000
Radiation Detectors	1.7	18
Total Power	105.94	

OPTIONS OF POWER SUBSYSTEMS

There are three basic categories from which to choose a source of power for the lunar lander. These categories include: 1. Carrying stored energy on board the spacecraft, 2. Gaining energy from the environment, and 3. A combination of carrying stored energy on board the spacecraft and gaining energy from the environment. By studying and understanding the functions and characteristics of the subsystems within each of these categories, a simple or complex design for the power subsystem can be designed.

The first category, carrying stored energy on board the spacecraft, is broken down into four subcategories. These subcategories include: Radioisotope Thermoelectric Generators (Static Nuclear Power), Primary Batteries, Fuel Cells, and Nuclear Reactors (Dynamic Nuclear Power).

The second category, gaining energy from the environment, consists of Solar Cells/Panels and Solar Dynamic Power. Solar Dynamic Power uses solar power to heat a working fluid to a vapor, which drives a turbine.

Finally, the third category, a combination of storing and gaining energy, is made up of Solar Cells and Secondary Batteries. These secondary batteries are unlike the primary batteries in that they are rechargeable.

The questions still remains - Which option will be able to satisfy the mission requirements specified on page 5-1? The best option can be found by going through the numerous pros and cons of each option. Figure 5-1 can narrow down the selection of a particular power subsystem quite quickly. This graph shows how electric power, in watts, is a function of the duration of use of the subsystem. For example, primary batteries can produce between 100 and 1000 watts of power; however, the power generated from these batteries will only last from 1 minute to a little over 1 day. Fuels cells become quite large as the duration of use increases; therefore, they are also not used. From category 1 there remains two choices, the RTG and the Nuclear Reactor. Nuclear power, although proven and effective, will not be used due to its ecological problems and many moving parts (due to dynamic nuclear power). The RTG is the only choice left. From Figure 5-1, RTGs do fit within the design specifications, and may be an option to explore further in this report.

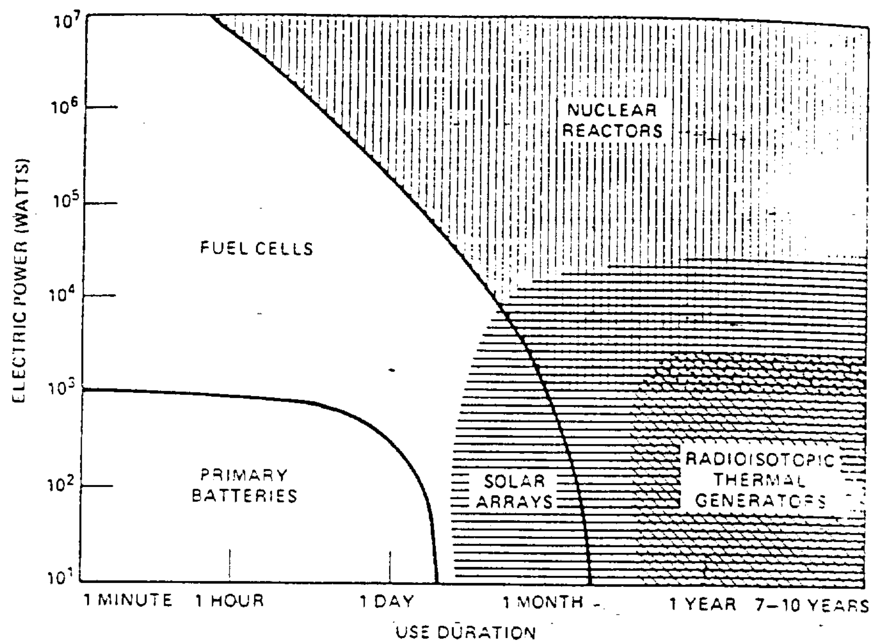


FIGURE 5-1

Although RTGs do fit our requirements for a long-term, lightweight, and compact power subsystem, there are a few other options that must be explored. The second category consists of solar cells/panels or solar dynamic power. Because of the 14-day lunar nights, once on the moon, using solar power exclusively, without some backup power mechanism, is not an efficient means of generating power for the probe; therefore, the second category is eliminated from consideration for a power subsystem.

Finally, the third option was the combination of secondary batteries with solar cells to power the spacecraft. If a battery could be designed to fit within the envelope of the lunar probe, this option might be feasible. The battery problem below rules out the possibly of this case, due to the large mass of both the NiCad and NiH_2 batteries.

THE BATTERY PROBLEM

Given: Lunar Lander must be provided with 150 Watts through 14-day Lunar Night (1 Eclipse per month)

Find: Total mass of Battery needed to accommodate the mission

Soln: For NiCad Battery

1. Find Maximum Eclipse Time

$$\text{Max } \textit{Eclipse Time} = 14 \textit{days } x \left(\frac{24\ hr}{1\ day}\right) = 336\ \textit{hours}$$

2. Find Stored Energy Needed

$$\textit{Stored Energy Needed} = (150\ W)(336\ \textit{hours}) = 50400\ Whr$$

3. Find Depth of Discharge

$$\text{Depth of Discharge} = (10 \text{ years})\left(12\ \frac{\text{eclipses}}{\text{year}}\right) = 120 \text{ cycles} => .85$$

4. Find Battery Capacity

$$\textit{Battery Capacity} = \left(\frac{50400\ Whr}{.85}\right) = 59294.12\ Whr$$

5. Find Mass (Without Packing Factor)

$$\textit{Mass} = \left(\frac{59294.12\ Whr}{25\ \frac{Whr}{kg}}\right) = 2372\ kg$$

6. Find Mass (With Packing Factor - 20% of Battery Mass)

$$\textit{Mass}_{tot} = 1.2(2372\ kg) = 2846\ kg$$

For NiH_2 Battery

$$\text{Mass}_{tot} = (1.2)\left(\frac{59294.12 \text{ Whr}}{45\ \frac{\text{Whr}}{kg}}\right) = 1581.2 \text{ kg}$$

As stated and seen above, the mass of a secondary battery is too large for the small lunar probe.

After exploring the numerous possibilities for a power subsystem for the lunar probe, the RTG was selected as the most feasible and best way to meet the requirements stated on page 5-1.

USING RTGS AS A SOURCE OF POWER

Before choosing which RTG best suits the mission requirements, it is important to understand the composition, advantages, and disadvantages of the RTG.

Currently, all U.S. spacecraft utilize Pu^{238} as a fuel source for their RTGs. Pu^{238}, which emits alpha particles, although poisonous to human beings, has very little effect on the spacecraft components. The alpha particles' low shielding requirements are necessary to the survivability of these critical spacecraft components over the required 10-year mission. The alpha particle isotopes do, however, give off Helium gas. This gas must be vented from the spacecraft.

There are many advantages of using an RTG as a source of power. These advantages must satisfy the mission requirements. They include:

- highly reliable over extended operating lifetimes (due to the long half life of Pu^{238} -> 87 years)
- compact
- rugged
- radiation resistant (unaffected by radiation effects encountered on lunar mission)
- easily adaptable to mission applications
- produce no noise, vibration, or torque during operation (vital to spacecraft components' survivability, and overall mission survivability)
- require no start-up devices to operate
- start producing electrical power for the payload as soon as the heat source is installed

- power output is easily regulated at design levels by maintaining a matched resistive load on the converter
- low to moderate weight
- safe and proven (on board many missions - See Table 5-2)

Although, as shown above, there are many advantages to using an RTG, there are some drawbacks to its use. The disadvantages include:

- costly
    * Pu^{238} costs approximately $3000 per Watt, as opposed to $2500 per Watt for Solar Photovoltaic Power, $800 per Watt for Solar Thermal Dynamic Power, and $400 per Watt for Nuclear Power.
- handling and safety procedures are complex and arduous
- workers must work with poisonous, radioactive material
- relatively low thermoelectric conversion efficiency, typically less than 10%; therefore, power subsystem must be integrated with the thermal control subsystem.

These disadvantages did not alter the lunar probe design, but were considered in its initial phases of design.

Now that the composition, advantages, and disadvantages of using a Pu^{238} fuel source RTG have been established, which RTG will best suit the mission requirements laid out on page 5-1? Over the years (from 1961 to present), RTG technology has changed. Examining Table 5-2, it can be seen that there are only two feasible choices to meet just the first requirement (Supply 150 Watts (BOL) power to the 10-year mission, with an EOL power of approximately 97 Watts). From Table 5-2, the Multi-Hundred Watt RTG and the General Purpose Heat Source RTG can both achieve this requirement.

TABLE 5-2 CHOICES OF RTG'S

Mission	SNAP-33 Early Transit 1961	SNAP-19 Pioneer (72,73) Viking (75,76)	TRANSIT RTG 1972	MHW Voyager (77)	GPHS RTG'S Galileo (90)
BOL Power (W)	2.7	28-43	36.8	150	285
Mass (kg)	2.1	13.6	13.5	38.5	55.9
Power Density (W/kg)	1.3	2.1-3.0	2.6	4.2	5.1
Efficency (%)	5.1	4.5-6.2	4.2	6.6	9

The GPHS-RTG was chosen to satisfy this long-term lunar mission. The following are the design specifications of this particular RTG.

GPHS-RTG SPECIFICATIONS

- BOL power: 285 W (with stack of 18 GPHS modules)

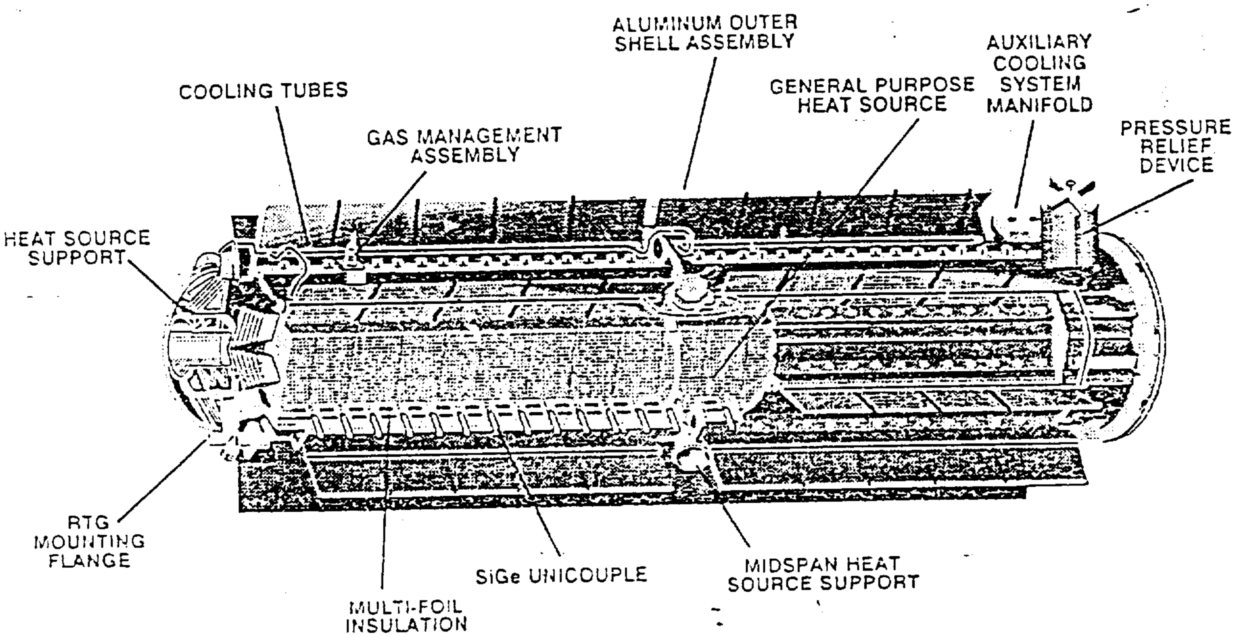
- operating voltage output: 30 volts DC

- Dimensions: 42.2 cm (16.6 in) diameter (fin tip to fin tip)
  114 cm (44.9 in) long

- Weight: 55.9 kg

- Specific power at launch: 5.1 W/kg

GENERAL PURPOSE HEAT SOURCE - RADIOISOTOPE THERMOELECTRIC GENERATOR



- POWER OUTPUT - 285 WATTS
- FUEL LOADING - 4400 WT; 132,500 Ci
- WEIGHT - 124 LBS
- SIZE - 16.6 IN × 44.5 IN

FIGURE 5-2

General Purpose Heat Source (GPHS)
Module Assembly

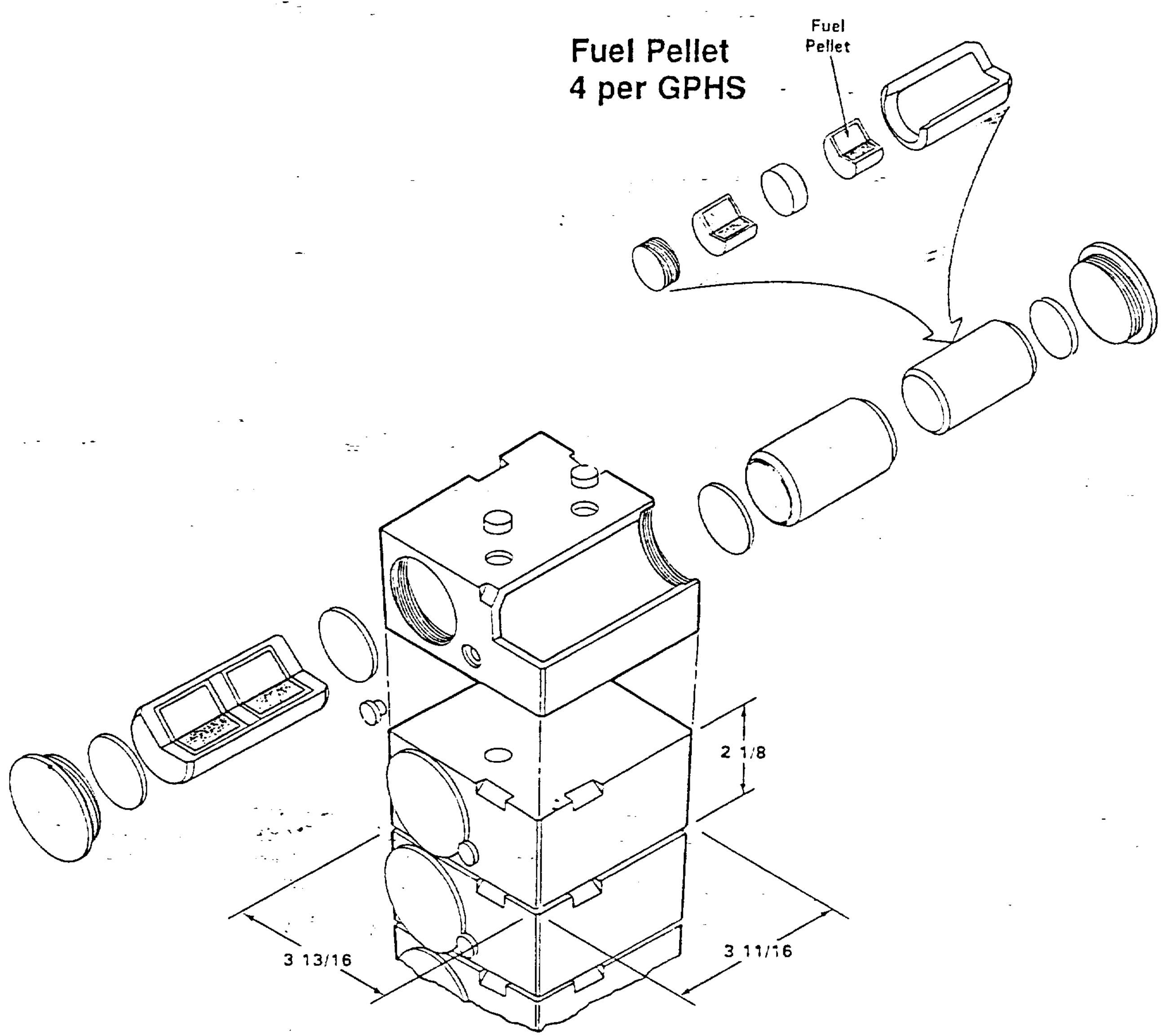


FIGURE 5-3

The GPHS-RTG, shown in Figure 5-2, was chosen because it is current state-of-the-art technology used in the United States' Space Program. Proven aboard the Galileo spacecraft, launched in October 1989, and the Ulysses spacecraft, launched in October 1990, this RTG is modular in design. This modular design allows alteration in the number of GPHS modules stacked within the whole unit, dependent on the power requirements. For instance, this mission requires a BOL power of 150 Watts. Since the GPHS-RTG can supply 285 Watts with 18 stacks, 18 stacks are not required. The number of stacks needed is:

$$\frac{150 \text{ Watts}}{285 \text{ Watts}} = \frac{\text{Num. stacks}}{18 \text{ stacks}} \Rightarrow 9.47 \text{ stacks} \Rightarrow \text{use 10 stacks}$$

Using 10 stacks would give a BOL power of 158 Watts. Figure 5-3 shows how the stacks, with Pu^{238} fuel, are arranged to fit within the rest of the RTG unit.

The EOL power can be interpolated from Figure 5-4.

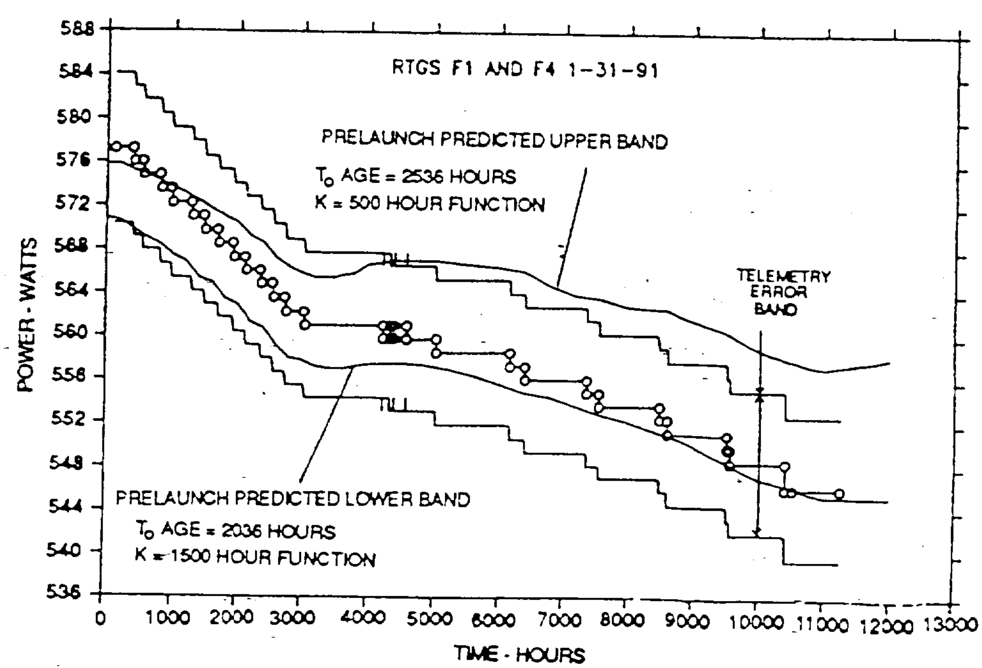


FIGURE 5-4

From FIGURE 5-4, Solve for K:

$$550 \text{ Watts} = 578 \text{ Watts } e^{-(10000 \text{ hours})(k)} \Rightarrow k = 4.96559 \times 10^{-6}$$

Using the 158 Watts BOL power, the EOL power is:

$$\text{EOL Power} = 158 \text{ Watts } e^{-(4.96559\times10^{-6})(87600 \text{ hours})} \Rightarrow 102 \text{ Watts}$$

where 87600 hrs = 10 yrs * 365 days * 24 hrs

Since the Attitude/Control subsystem will not be used once on the moon, the calculated EOL power will not be a problem. (see Power needed in Table 5-1)

To convert thermal energy from the decaying isotope into usable, electrical energy, the GPHS-RTG employs 572 SiGe thermocouples. Insulated by 60 alternating layers of 0.003 inch molybdenum foil and astroquartz cloth, these thermocouples (see Figure 5-5, next page) are connected in two series-parallel electric wiring circuits in parallel to increase reliability and provide full output voltage. The design of this circuit is such that the RTG is permitted to operate even if one unicouple becomes "shorted" or "opened". The thermocouples, therefore, are designed for longevity and reliability.

The SiGe thermocouples operate from a cold junction temperature of 573K to a hot junction temperature of 1273K. This range gives a thermoelectric efficiency of about 9 percent, with waste heat being radiated from the finned RTG housing. This housing is covered with a high emissivity coating. For this mission, the amount of thermal power supplied by the RTG is:

$$\frac{\textit{Thermal Energy}}{4410\ Wt} = \frac{10\ \textit{stacks}}{18\ \textit{stacks}} \Rightarrow T.E. = 2450\ Wt$$

Since the efficiency is only 9%, the thermal energy that must be dissipated is:

$$(2450\ Wt) - [(2450\ Wt)(.09)] = 2229.5\ Wt$$

This large amount of waste energy will make integration of the power subsystem with the thermal

control subsystem very critical. The actual methods for this waste disposal are discussed in the thermal control section, Chapter 7.

SILICON GERMANIUM UNICOUPLE

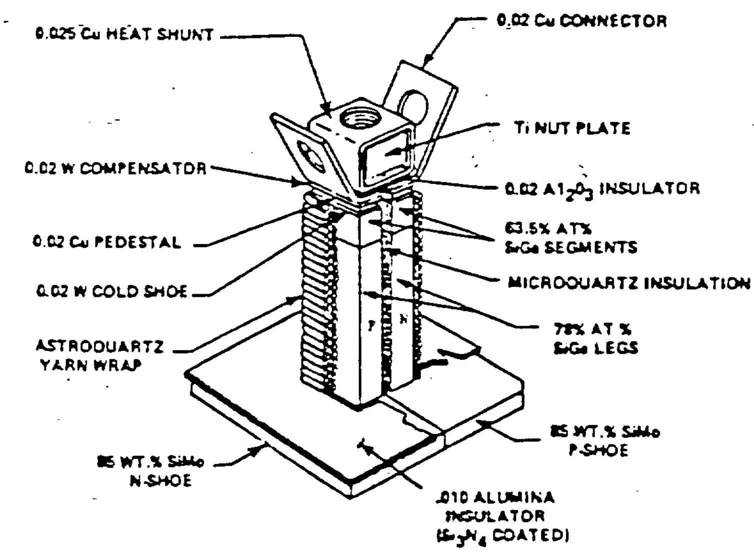


FIGURE 5-5

PROBLEMS ENCOUNTERED

The first problem encountered was in the high voltage required by the mass spectrometer. Because the mass spectrometer requires a 25000 volt (DC) input, the output voltage from the SiGe unicouples on the RTG must be stepped up from 30 Volts (DC) to 25000 volts (DC). The circuit, shown in Figure 5-6, describes just how this step-up will be accomplished. Basically, the step-up is a DC to AC conversion, then back to DC. This is accomplished by passing the signal through a transistor which uses a clock signal to simulate an AC output voltage given a DC input voltage, a transformer to amplify the voltage, and finally in parallel with a large capacitor to smooth out the signal and simulate a DC output.

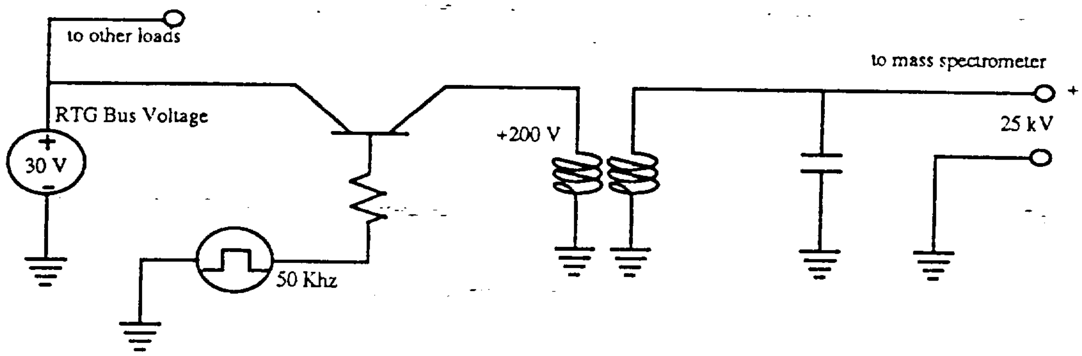


Figure 5-6 Mass Spectrometer Voltage Converter

The other problem encountered was that at various times large quick bursts of power need to be used to detonate small pyrotechnics (seismograph probe, exploding bolts for shroud, PAM-d attachment bolt release, and seismograph uncaging). To accomplish this task a large capacitor with a series of switches will be used for short term, high voltage / high power applications (see Figure 5-7).

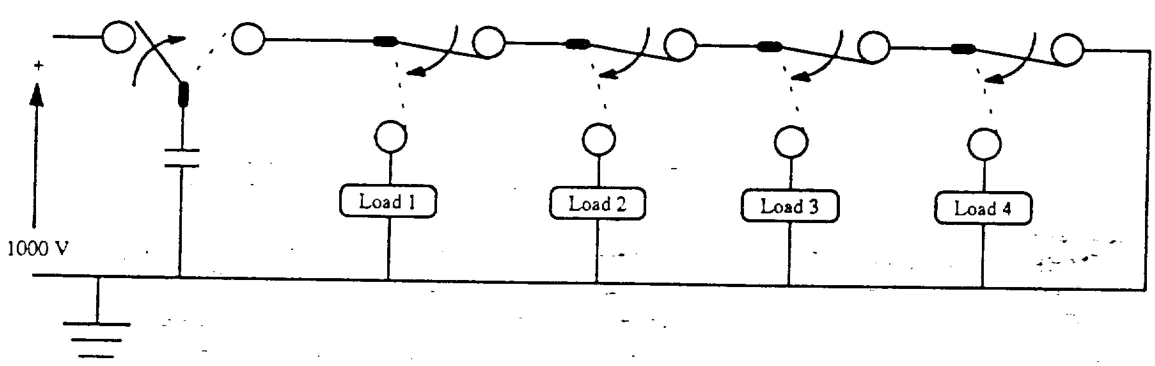


Figure 5-7 Short Term High Voltage / High Power Load System

CHAPTER 6

COMMUNICATIONS, CONTROL AND DATA HANDLING

CHAPTER 6 - COMMUNICATIONS

The Lunar Probe is going to have to talk with the earth while it is in flight and while it is on the moon. The dish on the probe is parabolic with a fifteen degree gimbled rotating antenna, and it will have an omnidirectional antenna as a backup. The gimbled antenna will cause it to be pointing at the earth constantly while it is in flight. The ground station, Wallops Island, will track the probe. Wallops is going to keep in touch with the probe during flight to check on the condition of the satellite and its position. Once on the moon the probe will communicate with the ground station once a day. The requirements of our dish on the satellite were made by data needed to be sent due to payload and spacecraft upkeep. Additionally, size restraints with structural design and compatibility with the ground station had to be considered.

The initial parameters include:

- frequency of transmission = 3 GHz
- Diameter of ground station dish = 18.3 m
- Efficiency of both dishes = .55
- Maximum distance between satellite and station =3.476 e8 m
- Link Margin = 3 dB
- Signal to Noise ratio for station = 10 dB
- Noise temperature of receivers = 1500 K
- Transmitter efficiency = 40 %
- Diameter of probe's dish = .9 m (due to thrust ring)

The link budget calculation is shown in Figure 6-1.

The bit rate that is used is 1.5 Mb/sec. This is the highest that Wallops can handle, and it was chosen so our satellite would only tie up Wallops' antenna for as little time as possible (maximum transmit time is 4.5 minutes). The spacecraft is receiving its data from the experiments onboard the probe and also from the subsystems. On board the probe is a seismograph, radiation

LINK BUDGET

$$G=\left(\frac{\pi D}{\lambda}\right)^2\eta$$

$$G_r=\left(\frac{\pi(18.288)}{.1}\right)^2(.55)=181549=52.29\text{ dB}$$

$$G_t=\left(\frac{\pi(.899)}{.1}\right)^2(.55)=429=26.32\text{ dB}$$

$$P_L=\left(\frac{4\pi R}{\lambda}\right)^2=\left(\frac{4\pi(4.063x10^8)}{.1}\right)^2=214.16\text{ dB}$$

$$T_L=214.16\text{ dB}+3\text{ dB}+2\text{ dB}=219.16\text{ dB}$$

$$N_0=\ kT=(1.38x10^{-23})(1500)=-196.84\frac{\text{dBW}}{\text{Hz}}$$

$$BW\ =\ 52.7\text{ dB-Hz}$$

$$P_r=\frac{S}{N}+BW+N_0=10+52.7-196.84=-134.1\text{dBW}$$

$$P_t=P_r+T_L+M-G_t-G_r$$

$$P_t=-134.1+219.16+3-26.32-52.29=9.15\text{ dB}$$

$$P_t=8.214\text{ Watts}$$

Since this is the minimum power needed to ensure transmission, we will use 25 Watts of power to transmit the data due to the 40% transmitter efficiency.

FIGURE 6-1

sensors, a mass spectrometer, all of which produce data to be stored for transmission. The probe will send this data down once a day, but the storage unit will be programmed to be able to hold the data for a maximum of thirty hours. The bit rate calculations are shown in Figure 6-2.

When the satellite is in view of Wallops, Wallops will send a signal to the probe to dump its data. The one gigabit storage unit will then dump all of the data into the 32 bit RISC processor (Figure 6-3) which will in turn send the data to Wallops. Wallops then routes the data

to the United States Naval Academy to be analyzed.

BIT RATES

Seismograph: $$\left(3000\frac{\text{bits}}{\text{sec}}\right)\left(60\frac{\text{sec}}{\text{min}}\right)\left(60\frac{\text{min}}{\text{hr}}\right)(30\text{ hr}) = 3.24\text{x}10^{8}\text{ bits}$$

Radiation Sensors: $$\left(640\frac{\text{bits}}{\text{sec}}\right)\left(60\frac{\text{sec}}{\text{min}}\right)\left(60\frac{\text{min}}{\text{hr}}\right)(30\text{ hr}) = 6.912\text{x}10^{7}\text{ bits}$$

Mass Spectrometer:
$$(62\text{ elements})\left(32\frac{\text{bits}}{\text{word}}\right)\left(1\frac{\text{word}}{\text{element}}\right)\left(60\frac{\text{samples}}{\text{hr}}\right)(30\text{ hr}) = 3.5712\text{x}10^{6}\text{ bits}$$

Spacecraft Housekeeping Data: $$\left(3000\ \frac{\text{bits}}{\text{hr}}\right)(30\text{ hr}) = 90000\text{ bits}$$

Total Storage Required: $= 3.97 \text{ x } 10^{8}$ bits

$= 397$ Mb

Usage: - 1 Gb Storage Unit

- 1.5 Mb/sec data transfer rate

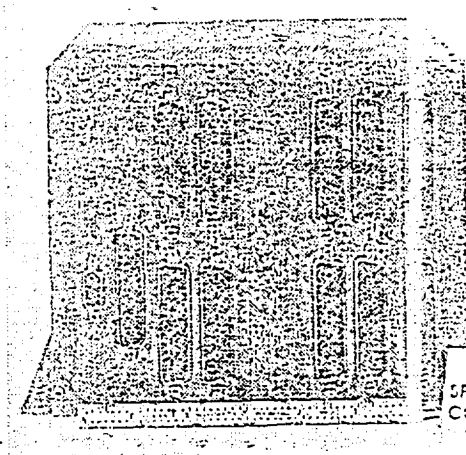
Transfer time: $$\left(\frac{397\text{ Mb}}{1.5\text{ Mb/sec}}\right) = 264.5\text{ sec} = 4\text{ min }24.5\text{ sec}$$

FIGURE 6-2

Spacecraft Controller

- Demonstrates Flight-Qualified MIL-STD-1750A & Full 32 Bit RISC Processor Architectures
  - First Flight Use Of 32 Bit RISC Processor
  - Multiple Processors Configured For Increased Fault Tolerance & Radiation Immunity
  - Upgradeable To RH3000 (20 MIPs / 7 MFLOPS) & RH6000 (40 MIPS / 20 MFLOPS)

Parameter	RH1750 Chipset	UTMC69 -R000 Processor	R3081 32 Bit RISC Processor
Command & Telemetry	Primary	-	Backup
Housekeeping	Primary	-	Backup
Data Formatting & Sequencing	Backup	Primary	Backup
Image Compression	-	Controls JPEG Chipset	Backup
Optical Sensor Controller	-	Primary	Backup
Image Processing	Star Tracker Only	-	Primary
MIPS	1.7	8.0	18
MFLOPS	0.7	0	3.5
Radiation Immunity (Rads (Si))	>100 k	>100 k	>15 k
Mass (g)	500	500	500
Volume (cm^3)	575	575	575
Comments	Advanced Mil-Std-1750A Architecture	16 Bit RISC Rad-Hard Micro Controller	Commercial 32 Bit RISC Processor



DSPSE SPACECRAFT CONTROLLER

FIGURE 6-3

CHAPTER 7

THERMAL DESIGN

CHAPTER 7 - THERMAL DESIGN

INITIAL CONSIDERATIONS

The thermal design of the lunar probe is based on the various systems on board and the temperature limits of these components. The maximum lunar lander temperature is the lowest of the maximum temperatures for each of the components and the minimum lander temperature is the highest of the component minimum temperatures. Table 7-1 shows the various operating temperatures of the spacecraft components and the resulting temperature band of 273 K to 313 K for the entire lander.

TABLE 7-1 TEMPERATURE LIMITS OF LANDER SYSTEMS

COMPONENT	MIN TEMP.	MAX TEMP.
Seismograph	263 K	328 K
Radiation Sensors	268 K	333 K
Mass Spectrometer	268 K	323 K
Electonics	250 K	328 K
Propellant	263 K	353 K
RTG	273 K	1273 K
OPERATING BAND	273 K	323 K

In order to minimize the complexity of the system, an entirely passive system is desired. If this is not possible, then a system with few moving parts and minimal power requirements is desired because of the long duration of the mission.

The thermal environments that the spacecraft must be designed for can be broken down into four parts: pre-launch, launch and transfer to moon, lunar day, and lunar night. Each of these phases has its own unique problems and solutions.

RTG waste heat must be transported to the lander outer surfaces so that it can be rejected

to the environment. This is accomplished through the use of heat pipes which transfer the high waste heat generated by the RTG to the outer panels for heat transfer.

PRE-LAUNCH

Prior to actual liftoff, the lander will be encased inside the Delta II shroud and will receive heat inputs from the following main sources: RTG waste heat, conduction from the Delta II, radiation from the Delta II shroud, and waste heat generated from the various electronic components on board. Of these inputs, the ones which will have the greatest effect will be the RTG waste heat (2200 W) and the conduction from the Delta II (based on Delta II temperature).

Two methods of removing the RTG waste heat are possible. Cooling coils are installed on the RTG for use if placed in the bay of a space shuttle. While the Delta II could be modified to facilitate some form of connected cooling system, the problem then arises when determining how to disconnect it from the RTG at launch and how the heat will be removed prior to shroud release. These reasons make this approach unfeasible. The second alternative is to run conditioned gas streams over the fins to cool the RTG. This system has been used on previous missions and has proven to be an effective means of removing RTG waste heat.

Delta II conduction will try to equalize the temperature between the Delta II rocket and the lunar probe. Since the Delta II has many components with approximately the same operating temperature bands that are on the lunar probe the Delta II should be maintained in band. Additionally, the operating band of the lunar lander is between 0°C and 40°C and practical engineering sense says that while parked on the launch pad the Delta II cannot exceed this temperature.

All other heat inputs should be minimal compared to those mentioned above, however,

accurate monitoring of the lander temperature should be maintained at all times. If the temperature begins to approach one of the operating limits then the gas flow to the RTG can be adjusted to either remove more or less heat from the lander as required. Since solar flux is non-existent during the fourteen day lunar night, the heat generated by the RTG must be used to maintain the temperature of the lander in band. This requires that the heat pipes be "turned off" to allow the heat from the RTG to warm the rest of the spacecraft and keep it in band. The specifics of the heat pipe design and how it will accomplish this task can be found later in this chapter.

LAUNCH AND TRANSFER

The next phase of the mission entails getting from Earth to the moon. During this portion the main thermal inputs are from solar flux, Earth or lunar albedo (depending on where the lander is in its transfer orbit), and waste heat generated by onboard systems (RTG and electronics). During this phase the spacecraft flies such that the top is pointed towards Earth, the bottom points toward the moon, and the sun strikes the side of the lander (temperature will remain within specifications as long as this profile is maintained within plus or minus 12°). As noted before, the payload section is covered during transit to shield the enclosed environment from solar radiation and to protect it from small particle damage. This protective covering will be covered with white epoxy to yield the IR emissivity and solar absorptivity found in Table 7-2. The sides and bottom must also be used during the lunar phase of the mission so the thermal characteristics of these sides must be able to keep the lander in operating limits in the presence of high lunar albedo and lunar infrared emissions. This requirement leads to covering the sides with multi layer radiative insulation which has an effective IR emissivity of 0.002 and a solar absorptivity of 0.080.

TABLE 7-2: THERMAL PERFORMANCE TRANSFER TO LUNAR ORBIT

Satellite Parameters:				
Surface Area Solar	5.120 m^2	5.120 m^2	5.120 m^2	5.120 m^2
Surface Area Earth	5.840 m^2	5.840 m^2	5.840 m^2	5.840 m^2
Surface Area Lunar	5.240 m^2	5.240 m^2	5.240 m^2	5.240 m^2
Total Surface Area	20.580 m^2	20.580 m^2	20.580 m^2	20.580 m^2
IR Emissivity: Sides	0.002	0.002	0.002	0.002
Solar Absorptivity: Sides	0.080	0.080	0.080	0.080
IR Emissivity: Top	0.888	0.888	0.888	0.888
Solar Absorptivity: Top	0.248	0.248	0.248	0.248
Flux Data				
Solar Flux	1374 W/m^2	1374 W/m^2	1374 W/m^2	1374 W/m^2
Earth IR Emissions (294 K)	258 W/m^2	258 W/m^2	258 W/m^2	258 W/m^2
Lunar IR Emissions (400 K)	871 W/m^2	871 W/m^2	871 W/m^2	871 W/m^2
Position Data				
Distance from Earth	120 km	1000 km	10000 km	50000 km
Radius of Earth	6378 km	6378 km	6378 km	6378 km
Angular Radius of Earth	1.378 rad	1.044 rad	0.400 rad	0.113 rad
Earth Albedo Correction	0.350	0.350	0.350	0.350
Distance from Moon	376164 km	375284 km	366284 km	326284 km
Radius of Moon	1738 km	1738 km	1738 km	1738 km
Angular Radius of Moon	0.005 rad	0.005 rad	0.005 rad	0.005 rad
Moon Albedo Correction	0.890	0.890	0.890	0.890
Thermal Inputs:				
RTG Waste Heat	2229.500 W	2230.000 W	2230.000 W	2230.000 W
Electronics Waste Heat	24.000 W	24.000 W	24.000 W	24.000 W
Solar Flux	562.790 W	562.790 W	562.790 W	562.790 W
Earth Emissons	2.437 W	1.499 W	0.238 W	0.019 W
Earth Albedo	181.053 W	110.243 W	14.897 W	1.039 W
Lunar Emissions	0.000 W	0.000 W	0.000 W	0.000 W
Lunar Albedo	0.004 W	0.004 W	0.004 W	0.005 W
TOTAL INPUT	2999.784 W	2928.536 W	2831.929 W	2817.854 W
Resulting Temperature	317.495 K	315.593 K	312.958 K	312. 68 K

Satellite Parameters:				
Surface Area Solar	5.120 m^2	5.120 m^2	5.120 m^2	5.120 m^2
Surface Area Earth	5.840 m^2	5.840 m^2	5.840 m^2	5.840 m^2
Surface Area Lunar	5.240 m^2	5.240 m^2	5.240 m^2	5.240 m^2
Total Surface Area	20.580 m^2	20.580 m^2	20.580 m^2	20.580 m^2
IR Emissivity: Sides	0.002	0.002	0.002	0.002
Solar Absorptivity: Sides	0.080	0.080	0.080	0.080
IR Emissivity: Top	0.888	0.888	0.888	0.888
Solar Absorptivity: Top	0.248	0.248	0.248	0.248
Flux Data				
Solar Flux	1374 W/m^2	1374 W/m^2	1374 W/m^2	1374 W/m^2
Earth IR Emissions (294 K)	258 W/m^2	258 W/m^2	258 W/m^2	258 W/m^2
Lunar IR Emissions (400 K)	871 W/m^2	871 W/m^2	871 W/m^2	871 W/m^2
Position Data				
Distance from Earth	100000 km	200000 km	300000 km	375284 km
Radius of Earth	6378 km	6378 km	6378 km	6378 km
Angular Radius of Earth	0.060 rad	0.031 rad	0.021 rad	0.017 rad
Earth Albedo Correction	0.350	0.350	0.350	0.350
Distance from Moon	276284 km	176284 km	76284 km	1000 km
Radius of Moon	1738 km	1738 km	1738 km	1738 km
Angular Radius of Moon	0.006 rad	0.010 rad	0.022 rad	0.688 rad
Moon Albedo Correction	0.890	0.890	0.890	0.890
Thermal Inputs:				
RTG Waste Heat	2230.000 W	2230.000 W	2230.000 W	2230.000 W
Electronics Waste Heat	24.000 W	24.000 W	24.000 W	24.000 W
Solar Flux	562.790 W	562.790 W	562.790 W	562.790 W
Earth Emissons	0.005 W	0.001 W	0.001 W	0.000 W
Earth Albedo	0.281 W	0.073 W	0.033 W	0.021 W
Lunar Emissions	0.000 W	0.000 W	0.002 W	2.075 W
Lunar Albedo	0.007 W	0.016 W	0.086 W	107.930 W
TOTAL INPUT	2817.083 W	2816.882 W	2816.912 W	2926.817 W
Resulting Temperature	312.547 K	312.541 K	312.542 K	315.547 K

With the solar flux striking a portion of the spacecraft, earth emissions and albedo affecting the top, and lunar emissions and albedo striking the bottom, a total environmental input can be determined. Along with RTG waste heat and various other electronic waste heat, this constitutes the entire thermal input to the spacecraft. Since heat output must equal heat input to maintain equilibrium the spacecraft will radiate the total heat input. The area for heat transfer includes all the sides, the top, and the bottom. Table 7-2 shows calculations for various points during the spacecraft's transfer orbit. Looking at the values for Earth albedo and emissions along the transfer, one can see that at around 50,000 kilometers from Earth, these inputs are negligible and can be discounted (this fact is used when determining thermal performance during the lunar phase of the mission). Another item to note is that the value given for Earth emissions is constant in Table 7-2 while it should decrease with increasing distance. The value given is the emission seen at 500 kilometers above the surface of the Earth. Since Earth emissions are only approximately 0.1% of total emissions at 120 kilometers above the surface this approximation does not considerably effect the overall thermal design of the spacecraft (the same is true for lunar emissions on the other end of the transfer orbit).

LUNAR SURFACE

Figures 7-1 and 7-2 show how the lunar surface temperature varies with sun angle and where the moon is in its diurnal cycle (data from Surveyer missions).

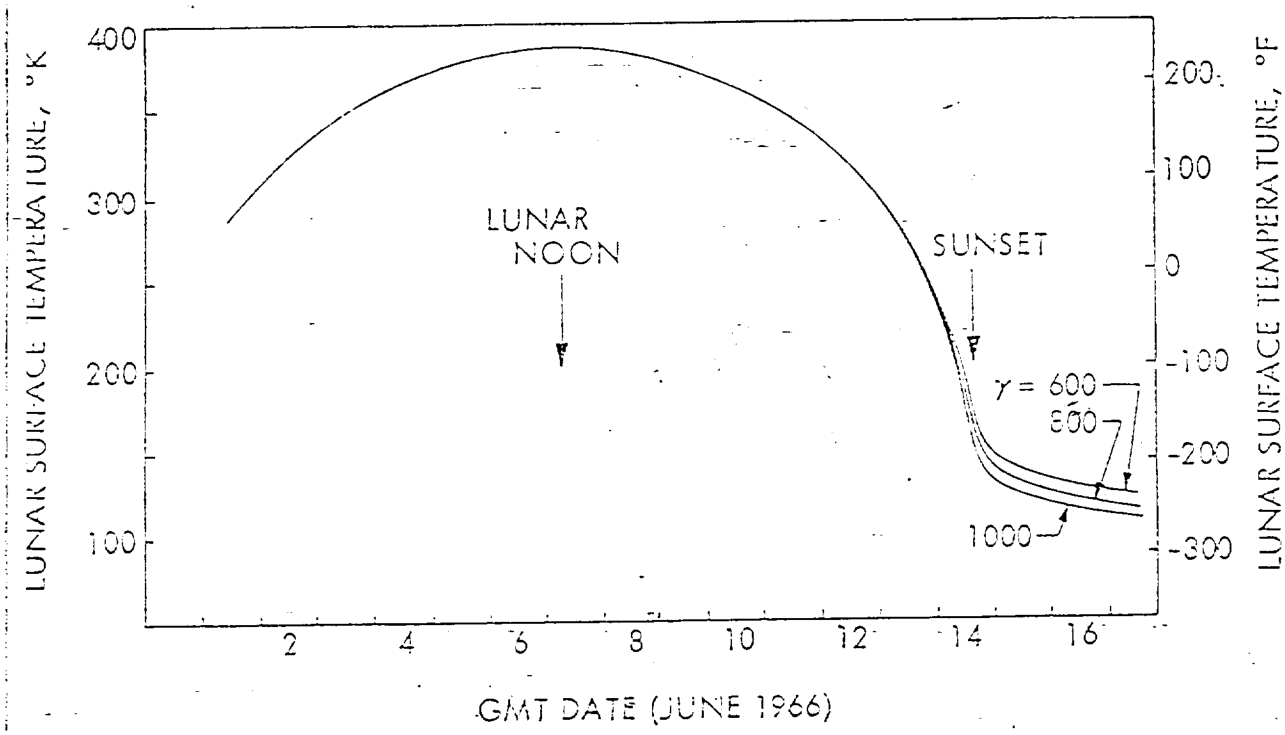


Figure 7-1 Variation of Lunar Surface Temperature with Diurnal Position

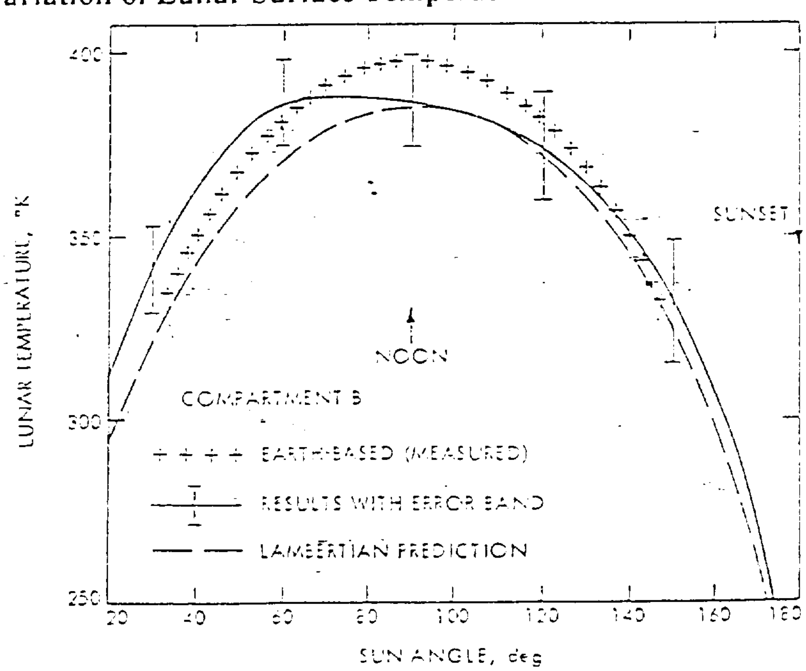


Figure 7-2 Variation of Lunar Surface Temperature with Diurnal Position

When the lander is in its final descent phase an explosive bolt is fired which releases the shroud covering the enclosed environment section. This shroud falls away to expose the environment section to solar flux. This enclosed environment consists of eight compartments with slightly different coverings. These coverings range from glass to boron impregnated plexiglass. The effective IR emissivity of the top section is 0.82 and a solar absorptivity of 0.081. On the lunar surface another explosive bolt is fired which drops the eight side panels down. This is done for two purposes: first, it increases the surface area for heat transfer so that during the hot portions of the lunar day the excess heat can be dissipated; and second, dropping the sides effectively shields the environment section from lunar emissions and albedo so that accurate readings can be made on solar flux and its effects on an enclosed environment. The inside portion of these panels are covered with white epoxy which has an IR emissivity of 0.888 and a solar absorptivity of 0.248 and the back sides of the panels (which face the lunar surface) are covered with the multi layer radiative insulation.

Table 7-3 shows the thermal performance of the lander as a function of sun angle. One thing to be taken into account is that at a sun angle of about 75 degrees the active control system of the heat pipes (described later) starts to take effect which partially insulates the lander structure from the RTG. This means that RTG heat transfer begins to become radiative and due to different surface areas the spacecraft temperature is maintained in band. During the fourteen day lunar night the primary source of heat input to the satellite is RTG waste heat (only 1.4% coming from lunar emissions and electronic waste heat).

TABLE 7-3: THERMAL PERFORMANCE ON LUNAR SURFACE

Satellite Parameters:				
Surface Area: Solar	25.780 m^2	25.780 m^2	25.780 m^2	25.780 m^2
Surface Area: Lunar	14.780 m^2	14.780 m^2	14.780 m^2	14.780 m^2
Total Surface Area	40.560 m^2	40.560 m^2	40.560 m^2	40.560 m^2
IR Emissivity: Sun View	0.888	0.888	0.888	0.888
IR Emissivity: Moon View	0.002	0.002	0.002	0.002
Solar Absorptivity: Sun View	0.248	0.248	0.248	0.248
Solar Absorptivity: Moon View	0.080	0.080	0.080	0.080
Flux Data:				
Solar Flux	1374 W/m^2	1374 W/m^2	1374 W/m^2	1374 W/m^2
Lunar IR Emissions	871 W/m^2	871 W/m^2	871 W/m^2	673 W/m^2
Position Data				
Lunar Surface Temperature	400 K	400 K	400 K	375 K
Solar Grazing Angle	0 deg	10 deg	20 deg	30 deg
Moon Albedo Correction	0.890	0.890	0.890	0.890
Thermal Inputs:				
RTG Waste Heat	2230.000 W	2230.000 W	2230.000 W	2230.000 W
Electronics Waste Heat	24.000 W	24.000 W	24.000 W	24.000 W
Solar Flux	8784.587 W	8651.129 W	8254.811 W	7607.675 W
Lunar Emissions	25.744 W	25.744 W	25.744 W	19.887 W
Lunar Albedo	361.477 W	599.494 W	776.359 W	855.265 W
TOTAL INPUT	11425.808 W	11530.367 W	11310.914 W	10736.827 W
Resulting Temperature:	306.205 K	306.903 K	305.432 K	301.480 K

Satellite Parameters:				
Surface Area: Solar	25.780 m^2	25.780 m^2	25.780 m^2	25.780 m^2
Surface Area: Lunar	14.780 m^2	14.780 m^2	14.780 m^2	14.780 m^2
Total Surface Area	40.560 m^2	40.560 m^2	40.560 m^2	40.560 m^2
IR Emissivity: Sun View	0.888	0.888	0.888	0.888
IR Emissivity: Moon View	0.002	0.002	0.002	0.002
Solar Absorptivity: Sun View	0.248	0.248	0.248	0.248
Solar Absorptivity: Moon View	0.080	0.080	0.080	0.080
Flux Data				
Solar Flux	1374 W/m^2	1374 W/m^2	1374 W/m^2	1374 W/m^2
Lunar IR Emissions	511 W/m^2	380 W/m^2	276 W/m^2	195 W/m^2
Position Data				
Lunar Surface Temperature	350 K	325 K	300 K	275 K
Solar Grazing Angle	40 deg	50 deg	60 deg	70 deg
Moon Albedo Correction	0.890	0.890	0.890	0.890
Thermal Inputs:				
RTG Waste Heat	2230.000 W	2230.000 W	2230.000 W	2230.000 W
Electronics Waste Heat	24.000 W	24.000 W	24.000 W	24.000 W
Solar Flux	6729.384 W	5646.623 W	4392.293 W	3004.506 W
Lunar Emissions	15.091 W	11.219 W	8.146 W	5.751 W
Lunar Albedo	822.310 W	690.000 W	493.787 W	282.572 W
TOTAL INPUT	9820.784 W	8601.843 W	7148.226 W	5546.828 W
Resulting Temperature:	294.833 K	292.021 K	290.687 K	29[illegible]349 K

Satellite Parameters:			
Surface Area: Solar	25.780 m^2	25.780 m^2	25.780 m^2
Surface Area: Lunar	14.780 m^2	14.780 m^2	14.780 m^2
Total Surface Area	40.560 m^2	40.560 m^2	40.560 m^2
IR Emissivity: Sun View	0.888	0.888	0.888
IR Emissivity: Moon View	0.002	0.002	0.002
Solar Absorptivity: Sun View	0.248	0.248	0.248
Solar Absorptivity: Moon View	0.080	0.080	0.080
Flux Data			
Solar Flux	1374 W/m^2	1374 W/m^2	1374 W/m^2
Lunar IR Emissions	133 W/m^2	54 W/m^2	3 W/m^2
Position Data			
Lunar Surface Temperature	250 K	200 K	100 K
Solar Grazing Angle	80 deg	90 deg	>90 deg
Moon Albedo Correction	0.890	0.890	0.890
Thermal Inputs:			
RTG Waste Heat	2230.000 W	2230.000 W	2230.000 W
Electronics Waste Heat	24.000 W	24.000 W	24.000 W
Solar Flux	1525.427 W	0.000 W	0.000 W
Lunar Emissions	3.928 W	1.609 W	0.101 W
Lunar Albedo	105.707 W	0.000 W	0.000 W
TOTAL INPUT	3889.063 W	2255.609 W	2254.101 W
Resulting Temperature:	292.079 K	295.791 K	295.742 K

HEAT PIPE DESIGN

During the lunar day it is necessary to move the heat generated by the RTG from the center of the spacecraft to the side panels so that it may be dissipated and the lander kept below maximum temperature limits. However during the cold lunar nights the heat generated by the RTG must be retained by the lander to prevent the lander from dropping below minimum temperature requirements. This is accomplished by using variable conductance heat pipes made from aluminum. The specific types of heat pipes used are gas-loaded heat pipes with feedback controlled reservoirs. Use of a 1.93 cm diameter aluminum pipe, a wrapped screen wicking material, and ammonia as the working fluid will allow the lander to be maintained within the temperature band imposed by the various spacecraft components. Previous heat pipe designs have been used where a non-condensable gas is used in the reservoir to control vapor flow in the heat pipe, however, new research shows that use of the working fluid in conjunction with a

special baffle system between the heat pipe and the reservoir will accomplish the same task. These pipes are capable of carrying 456 W-m with a temperature differential from the condenser end to the evaporator end of 1.3°C. Since the distance the heat must be carried is 0.8 meters this means that each pipe will be capable of transporting 570 Watts of waste heat at maximum operating efficiency. Thus, since a maximum of 2230 Watts of heat must be transported, a total of four heat pipes must be used.

Figure 7-3 shows what the basic heat pipe looks like:

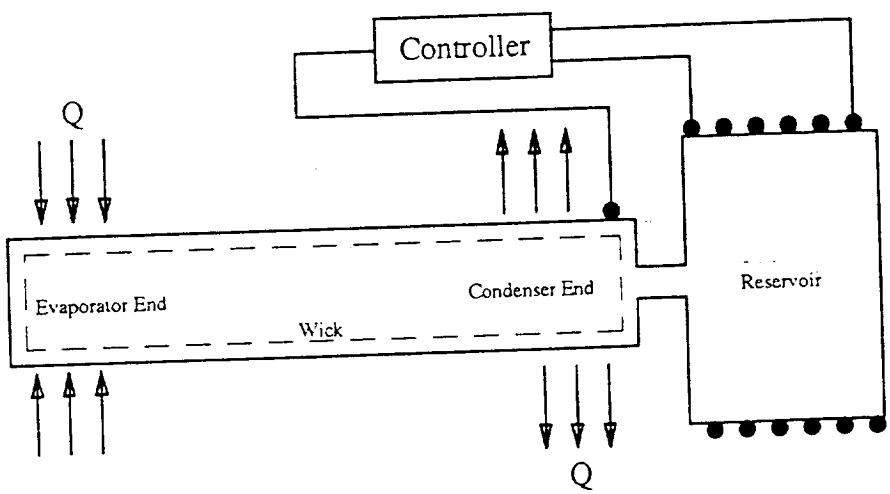


Figure 7-3 Basic Heat Pipe Schematic

The controller maintains a constant temperature on the condenser end by varying the vapor pressure of the working fluid. If the temperature on the condenser end begins to drop then the controller sends more current to heater coils at the reservoir. This causes a higher vapor pressure to be experienced inside the heat pipe which limits vapor flow from the evaporator to the condenser. This means that the operating temperature of the fluid is closer to the source temperature and heat is trapped inside the spacecraft. Conversely, as the condenser temperature rises the controller sends less current to the heater coils, reducing the vapor pressure, and

allowing for a greater amount of heat flow to the exterior surfaces. Total power required by the controller and heater assemblies is 10 Watts with a total mass for the four pipes of 6.8 kg.

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